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GENERAL ENVIRONMENTAL TEST SPECIFICATION FOR SPACECRAFT AND COMPONENTS

Goddard Space Flight Center
Greenbelt, MD

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GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND

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GENERAL

ENVIRONMENTAL TEST SPECIFICATION

FOR

SPACECRAFT AND COMPONENTS

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Date

GODDARD SPACE FLIGHT CENTER

GREENBELT, MARYLAND

FOREWORD

This document is based on the combined knowledge and experience accumulated since 1960 at the Goddard Space Flight Center in the development, environmental testing, and flight of a wide variety of spacecraft. All previously issued general environmental test specifications for GSFC spacecraft are updated and consolidated in this one document, and data have been provided for the testing of spacecraft to be launched by Titan and Atlas-Centaur launch vehicles recently adopted for GSFC missions. The specification has undergone Centerwide review and was approved by the GSFC Reliability Assurance Council in February 1969.

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ORGANIZATION OF THIS DOCUMENT

The specification consists of four sections and five appendixes as follows:

SECTION 1 - General information on testing; special problem areas, such as flight spares; mandatory and optional tests; failures; test facilities; and transportation and handling of spacecraft

SECTION 2 - Basic tests for all GSFC spacecraft (test levels that vary with the launch vehicle are carried in separate appendixes as described below); the section is subdivided as follows:

2.1 Spacecraft Design Qualification
2.2 Spacecraft Flight Acceptance

SECTION 3 - Basic tests for components, including subsystems and experiments; the section is subdivided as follows:

3.1 Component Design Qualification
3.2 Component Flight Acceptance

SECTION 4 - Terms and concepts defined according to their special usage in the environmental testing of spacecraft

LAUNCH VEHICLE APPENDIXES - The spacecraft test levels and durations which vary with the launch vehicle are carried in separate appendixes. The appendixes and the main text constitute the complete environmental test program

Appendix A - Delta L, M and N
Appendix B - Atlas-Centaur
Appendix C - Titan III-C
Appendix D - Thorad-Agena
Appendix E - Scout

SPECIAL NOTE: All appendixes were not available at the time of first printing. As they are published they will be mailed to users whose names appear on the distribution list per paragraph 1.3.

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TAB TITLES

For easy use of the document, the user is encouraged to cut and attach tab titles to section pages.

Contents	Definitions
Gen. Info.	Appen A - Delta
S/C Design Qual.	Appen B - Atlas-Centaur
S/C Flight Accep.	Appen C - Titan
Component Design Qual.	Appen D - Thorad-Agena
Component Flight Accep.	Appen E - Scout

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SECTION 1

GENERAL INFORMATION

SECTION 1

GENERAL INFORMATION

1.1 APPLICATION. This specification, S-320-G-1, covers environmental testing of spacecraft and components launched by the Delta, Atlas-Centaur, Titan III-C, Thorad-Agena, and Scout launch vehicles.

It supersedes the Delta, Thor-Agena, and Scout specifications (S-320-D-2, S-320-A-1, and S-320-S-1 respectively) and adds environmental test programs for spacecraft/components launched by Atlas-Centaur and Titan III-C launch vehicles.

1.2 PURPOSE. S-320-G-1 provides project managers with source material and a model format for the preparation of particular environmental test specifications for individual spacecraft projects.

A series of environmental tests are presented, many of which normally constitute required test programs for spacecraft and components. Some spacecraft may have special characteristics, however, which require additional tests or which permit the deletion of some of the tests. (See 1.6.4, 1.6.5 and 1.6.6 for listing of mandatory, optional and special tests.)

1.3 DISTRIBUTION OF REVISIONS. The document will require additions or revisions as the knowledge of the space and launch environment is increased and testing technology improves, or as new launch vehicles are adopted for GSFC projects. Recipients of original distribution will automatically receive revisions. Others who wish to receive them should request inclusion on the distribution list in a memorandum to Research and Technology Office, Code 327, Test and Evaluation Division, GSFC, Greenbelt, Md., 20771.

1.4 RESPONSIBILITY FOR ADMINISTRATION. The responsibility and authority for decisions in connection with the applicability of the requirements of this specification rest with the project manager, subject to review by the Reliability Assurance Council, GSFC.

1.5 APPLICABLE DOCUMENTS

1.5.1 Test Specifications and Reliability Plan. Spacecraft and component test specifications for a particular project shall be developed for conduct of the project's environmental test program. A Reliability Assurance Program Plan shall also be developed for each spacecraft project.

1.5.2 Launch Vehicle Reference Documents

(a) Delta Design Restraints Manual. (SM-48897), January 1966, and all most recent changes

(b) The Delta Launch Vehicle Powered Flight Dynamic Environment (Delta Vehicle Models L, M & N), Report No. DR 109, May 1968.

(c) Scout User's Manual (Ling-Temco-Vought Corp., Dallas, Texas), September 1965, and all most recent changes

(d) Centaur Payload Users Manual (NASA CR-72109) August 1966, in accordance with all most recent changes

(e) Titan III C Payload Users Guide (MCR-68-62), Martin Marietta Corp., October 1968

(f) Agena D Mission Capabilities and Restraints Catalog, Vol II, (TNX-521 46) Lewis Research Center, Cleveland, Ohio, November 1965, and all most recent changes.

1.5.3 Documents for General Information. These documents are listed for general information except item (e) which is needed for conduct of electromagnetic interference testing. All are listed with the understanding that they will be used in accordance with latest revisions.

(a) MIL-STD-810B, Environmental Test Methods, June 1967

(b) MIL-C-45662, Revision A, Calibration Systems Standards

(c) Standard Laboratory Information Manual, August 31, 1964 (Metrology Engineering Center Laboratory, General Dynamics Corp., Pomona, California)

(d) MIL-D-9412D, Data for Aerospace Ground Equipment

(e) Electromagnetic Compatibility (MSFC - SPEC - 279, June 1, 1964)

(f) Electromagnetic Interference Test Requirements and Test Methods (MIL-STD-826), USAF, January 20, 1964.

1.6 THE TEST PROGRAM. The specification is based on current scientific knowledge and experience gained at GSFC since 1960 in the environmental testing of spacecraft and components.

The following paragraphs under 1.6 provide basic information and a general outline of the design qualification and flight acceptance test programs.

1.6.1 Electrical Performance Record

1.6.1.1 Initial Test. Prior to the performance of any of the environmental test programs specified herein, the spacecraft/component shall be subjected to a comprehensive operational checkout in accordance with 2.1.3, 2.2.2, 3.1.4, or 3.2.2 as applicable and under the conditions specified in paragraph 1.9.4. A record shall be made of all data necessary to determine that performance of the specimen complies with the requirements of the particular spacecraft specification. The data provides a basis for checking satisfactory performance of the spacecraft/component before, during, or after environmental tests.

1.6.1.2 Succeeding Tests. The responsible test engineer shall maintain a chronological log of all periods of electrical performance including the Initial Test (1.6.1.1). Besides showing the duration of each operational period, the log shall show for any particular time the spacecraft/component operational mode and applicable mechanical configuration as well as the environmental exposure and stimulus being applied.

1.6.2 Installation Check. Following installation in the test apparatus and prior to exposure, the specimen shall be operated to insure that no malfunction or damage was caused due to faulty installation procedure or handling.

1.6.3 Evaluation of Spacecraft Performance. As directed in individual test procedures, a spacecraft undergoing tests shall be operated for evidence of deterioration to permit the collection of performance data. Provision shall be included when feasible for checkout of redundant subsystems and, where applicable, redundant components of circuitry at the appropriate level of testing to verify satisfactory performance.

1.6.4 Spacecraft Design Qualification

1.6.4.1 Mandatory and Optional Tests. The tests printed in capital letters in 1.6.4.2 are mandatory and may not be omitted without prior approval by the Assistant Director for Systems Reliability, GSFC. The other tests are optional but are recommended for careful consideration depending upon the design and mission of the spacecraft.

As indicated in the applicable sections, the mandatory as well as the optional tests can often be satisfied in alternative ways depending on the needs of the project. For example, the operational spin and mechanical functioning test may satisfy the requirements of the shock test; and tests on engineering units combined with analysis may meet the requirements of structural loads, thermal balance, and operational spin and mechanical functioning. When option is permitted, the particular spacecraft specification shall state the selected option.

1.6.4.2 Test Sequence. The sequence recommended here is not obligatory, but changes to it shall not adversely affect the overall validity of the test program.

a. Initial Magnetic Field Measurement

b. Leak Detection*

c. ELECTRICAL PERFORMANCE**

d. Pyrotechnic Performance

e. Balance

f. Spin

g. Physical Measurements (Weight, Center of Gravity, and Moments of Inertia)

h. Temperature and Humidity

i. VIBRATION

j. Acoustic Noise

k. SHOCK

l. STRUCTURAL LOADS OR ACCELERATION, STEADY STATE

m. THERMAL-VACUUM PERFORMANCE

n. THERMAL BALANCE

o. Antenna Pattern Determination

p. Electromagnetic Interference

*In addition to the initial test, it is desirable to repeat the leak test before and after the temperature and humidity test and before and after the vibration and acoustic noise tests.

**In addition to the initial test, the electrical performance test should be conducted before, during and after each exposure in the spin, temperature, humidity, vibration, acceleration, thermal-vacuum and thermal balance tests.

q. OPERATIONAL SPIN AND MECHANICAL FUNCTIONING

r. Final Magnetic Field Measurement

1.6.5 Spacecraft Flight Acceptance

1.6.5.1 Mandatory and Optional Tests. The tests printed in capital letters in 1.6.5.2 are mandatory and may not be omitted without prior approval of the Assistant Director for Systems Reliability, GSFC. Other tests are optional but are recommended for careful consideration depending upon the design and mission of the spacecraft.

1.6.5.2 Test Sequence. The sequence recommended here is not obligatory, but changes to it shall not adversely affect the overall validity of the test program.

a. Leak Detection*

b. ELECTRICAL PERFORMANCE**

c. Pyrotechnic Performance

d. Balance - Initial

e. Physical Measurements (Weight, Center of Gravity, and Moments of Inertia)

f. Spin

g. VIBRATION

h. Acoustic Noise

*In addition to the initial test, it is desirable to repeat the leak test before and after the vibration and acoustic tests.

**Besides the initial electrical performance test, the test should be conducted before, during and after each exposure in the spin, vibration/acoustic noise, and thermal vacuum/thermal radiation tests.

i. SHOCK (May be Satisfied under Operational Spin and Mechanical Functioning)

j. SPACE ENVIRONMENT OPERATIONS CHECK

k. Final Balance

l. Antenna Pattern Determination

m. Electromagnetic Interference

n. OPERATIONAL SPIN AND MECHANICAL FUNCTIONING

o. Magnetic Field Measurement

1.6.6 Special Tests. The tests listed in 1.6.4 and 1.6.5 constitute the basic test program for prototype and flight spacecraft, respectively. Other tests such as boost heating simulation, boost pressure profile (venting), system life test, proof testing of high pressure systems, and ordnance safety shall be performed as deemed necessary. The test specifications, procedures, and decision criteria for any special test requirement shall be included in the test specification prepared for that spacecraft.

1.6.7 Criteria for Unsatisfactory Performance or Construction. Deterioration or change in performance of any component which could or does in any manner prevent the spacecraft from meeting functional, operational, or design requirements throughout the specified life shall provide reason to consider the spacecraft as having failed to comply with the conditions of the test to which it was subjected and shall be interpreted as a discrepancy.

1.6.8 Failure and Retest

1.6.8.1 In-line Failures. If an in-line failure occurs (e.g., a failure in the data transmission link) during a test, the test shall be discontinued. After corrective action has been completed, the test in which the failure occurred shall be repeated in its entirety without a failure before proceeding to the next test unless otherwise specified by the GSFC project manager.

1.6.8.2 Failures With Retroactive Effect. If corrective action taken as a result of failure (e.g., redesign of a component) affects the validity of previously completed tests, all such prior tests shall be repeated, unless otherwise specified by the GSFC project manager.

1.6.8.3 Failures With Limited Effect. Should a failure occur which has a limited effect on the overall spacecraft or subsystem, the project manager or his representative shall determine the feasibility and value of continuing the test to its specified conclusion before corrective action is undertaken.

1.6.8.4 Failure Reporting. Every failure shall be noted and reported in accordance with the provisions of a discrepancy reporting system meeting the requirements of the project Reliability Assurance Program Plan (RAPP).

1.7 SUBSTITUTION OF COMPONENTS

1.7.1 Conditions of Substitution. If a component or subassembly is operated in excess of design life and wears out or becomes unsuitable for further testing during a test sequence due to causes other than design deficiencies, a different component or piece of equipment may be substituted. If, however, the substitution substantially affects the significance of results of the test sequence during which the part failed, that test sequence and any previously completed procedures which are affected shall be repeated.

1.7.2 Replacement of "One Shot" Components. If a component or subassembly is expended during test as a normal consequence of irreversible or "one shot" operation, the component or subassembly shall be replaced with one that has met the required quality control standards or auxiliary tests for such components. A level of 3 sigma reliability is recommended. Examples are pyrotechnic devices, yo-yo despin weights, and elements which absorb impact energy by plastic yielding.

1.8 SEPARATE TESTING OF COMPONENTS

1.8.1 Purpose of Separate Tests. Components become design qualified only after successfully undergoing design qualification tests as part of an integrated spacecraft system. However, most components are tested prior to integration in the prototype spacecraft for two reasons: (1) to reduce the chance of failure during spacecraft design qualification tests when such failure would seriously jeopardize the project schedule, and (2) to demonstrate conformance with procurement contracts when components are being purchased from another organization.

1.8.2 Waiver of Component Tests. The separate testing of components as set forth in Section 3 of this specification may be waived by the project manager.

1.8.3 Testing of Flight Spares

1.8.3.1 The Need for Spares. Goddard Space Flight Center has endorsed the full-systems test approach, in which the entire system is tested under conditions as realistic as possible. However, it is necessary to have a supply of tested replacement components or "spares" in case of component failure on the flight spacecraft during that period before launch and after systems tests have been completed. For that reason careful consideration must be given to the separate testing of spares. See 4.18 for definition of spare.

1.8.3.2 Testing Levels. Under conditions of 3.1.6.3 spare components may be subjected to design qualification test levels and flight acceptance durations. Otherwise spares are tested under flight acceptance for components per 3.2.

1.8.3.3 Major Considerations in the Testing of Spares. The test history of the component selected as a flight spare is the major consideration in deciding on the need and extent of further testing. If the spare is selected from a prototype or backup model spacecraft that has undergone system tests, it can be used with a high degree of confidence and little or no further testing may be needed;

otherwise it must be tested separately at the flight acceptance level for components per Section 3.

If a flight component is replaced for reasons of failure and it is then repaired and redesignated as a spare component, it shall be retested according to the provisions of 1.6.8.

When spares are tested as separate units it is particularly important to simulate as closely as possible the testing imparted to integrated flight components.

1.8.3.4 Caution on the Use of Spares. When the need for a spare arises, immediate analysis and review of the failed flight component must be made. If failure occurs in a component of which there are others of identical design, the fault may prove to be generic, affecting all components of that design, including the spare. In that case the testing of the spare may have to be extended to gain assurance of its flight readiness.

1.8.3.5 Spares for Proto-Flight Spacecraft. (See 4.11 for definition of proto-flight.) Testing of spares for proto-flights is normally prescribed under 3.2, component flight acceptance; however, should the spares become available for testing before the proto-flight components, they are tested at design qualification levels using flight acceptance durations (see 3.1.6.3). The proto-flight components would then be tested at flight acceptance levels and durations per 3.2 before being integrated into the spacecraft.

1.9 TEST FACILITIES

1.9.1 General. The apparatus used in conducting tests shall be capable of producing and maintaining the test conditions required with the specimen under test installed on the apparatus and operating or not operating as required.

1.9.2 Volume. The volume of the test facilities shall be such that the bulk of the specimen under test shall not interfere with the generation and maintenance of test conditions.

1.9.3 Heat Source. The heat source of the test apparatus shall be so located that radiant heat shall not fall directly on the specimen under test, except where application of radiant heat is one of the test conditions.

1.9.4 Standard Conditions for Test Area. Laboratory conditions for conducting specimen operational checkout prior to or after an environmental exposure shall be as indicated below, unless the specimen is sealed, protected, or otherwise functionally insensitive to variation in temperature and humidity. In those cases, checkout at room ambient conditions shall be acceptable.

a. Temperature: 25° ± 3°C

b. Relative humidity: 55 percent or less

c. Barometric pressure: Room ambient. (If so specified in the particular spacecraft specification, the performance data shall be corrected to 760 torr (29.98 in. Hg).)

1.10 MEASUREMENTS AND TOLERANCES

All measurements shall be made with calibrated instruments which are appropriate for the environmental conditions concerned.

The maximum allowable tolerances for test conditions and physical properties shall be as follows, unless otherwise specified by the applicable test section in the particular spacecraft test specifications. The values are exclusive of instrument accuracy.

a. Weight:
   0 to 100 lbs: ±.25 lb.
   more than 100 lbs: ±0.25%

b. Temperature: ±1°C (on controlled temperature sensors).

c. Stabilized Temperature: Temperature sensors vary less than 0.5°C per hour for a period of three hours.

d. Humidity: +0, -5% R.H.

e. Vibration amplitude:
   Sinusoidal: ±10%
   Random (Overall RMS level): ±10%
   Random (Power Spectral Density): ±3 db

f. Vibration frequency: ±2% or 1 Hz
   (whichever greater)

g. Acceleration: +0%, -5% (Paragraph 2.1.12.8)

h. Zero gravity simulation: In general, the effect of inadequate gravity compensation on load or dynamics shall be as low as necessary to achieve the test objective. As a guide, the g-effect shall be less than 10% of the operational loads. A residual of ±0.1 g is both achievable and acceptable for stage separation tests and for comparative measurements of appendage positioning provided the "sign" is correct (the net shear and moment imposed during measurement acts in the same direction as it would in flight, thereby causing any mechanism with backlash to assume the correct extreme position). For testing of mechanical functions such as appendage deployment, ±0.05 g is usually required.

i. Mechanical shock -
   Response spectrum: +50%, -10%

j. Additional tolerances: as specified in particular spacecraft specifications.

1.11 TRANSPORTATION AND HANDLING

To insure that environmental conditions resulting from transportation and handling do not exceed the levels imposed by the tests, thereby imposing unnecessary penalty on the design of spacecraft, the shipping and handling environment shall be controlled by specified modes of transportation, handling, and by the use of properly designed shipping containers.

SECTION 2

TESTING OF SPACECRAFT

2.1 SPACECRAFT DESIGN QUALIFICATION

2.2 SPACECRAFT FLIGHT ACCEPTANCE

2.1 SPACECRAFT DESIGN QUALIFICATION

The purpose of the design qualification program is to demonstrate the ability of the prototype spacecraft to meet all performance requirements and suffer no harmful degradation when exposed to environments considerably more stringent than those expected from flight-acceptance testing, prelaunch, launch, injection and orbit.

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2.1.1 Initial Magnetic Field Measurement

2.1.1.1 General Requirements. Spacecraft shall be subjected to a magnetic field measurement as required by the detailed spacecraft specification to determine the permanent, induced, and stray magnetic moments of the speacecraft. In general, spacecraft which carry magnetometers must receive much more meticulous magnetic inspection, to more stringent limits than spacecraft without such equipment. This assures that the onboard magnetic disturbance will not interfere with the accurate determination of the magnetic field in space. Also, the magnitude of the net magnetic moment of spacecraft must be determined to allow prediction of the change in spacecraft attitude caused by magnetic torque.

2.1.1.2 Setup. The spacecraft shall be positioned inside the magnetic test coils via a fixture of nonmagnetic material so that the magnetic properties of the spacecraft can be accurately measured.

2.1.1.3 Limits and Procedures. Limits and procedures of this test shall be prescribed by the particular spacecraft specification based upon consideration of spacecraft configuration and mission requirements. In some cases a deperm treatment may be required to reduce the moment caused by permanent magnetization to acceptable design limits.

2.1.2 Leak Detection

2.1.2.1 Applicability. Spacecraft which operate as hermetically sealed units or which have components/subsystems/experiments which operate as hermetically sealed units shall be subjected to a leak check. Tables I and II contain the applicable parameters.

2.1.2.2 Time of Performance. In addition to the initial leak check, the test should be repeated before and after the temperature and humidity test phase and before and after the vibration and acoustic noise test phase. The final leak test may be accomplished as part of the thermal-vacuum test.

TABLE I

LEAK DETECTION VACUUM TEST SCHEDULE
SPACECRAFT DESIGN QUALIFICATION

Applicable to	Pressure of Sealed Specimen	Proportion of Tracer Gas (Pressure)	Chamber Pressure	Maximum Leak Rate*
Sealed Spacecraft or Spacecraft with Sealed Units	100 to 760 torr	100%	1×10^{-4} torr	1×10^{-6} atm std. cc/sec.
	760 to 1520 torr (Absolute)	10 to 100%		

*Or as otherwise established by spacecraft design. Leak rate of pressurized gas attitude control systems during nonoperating phases of the operational cycle shall not exceed the maximum leak rate established by design limits.

TABLE II

LEAK DETECTION NON-CHAMBER "SNIFF" TEST
SPACECRAFT DESIGN QUALIFICATION

Applicable as Required To	Pressure of Sealed Specimen (Induced)	Proportion of Tracer Gas (Pressure)	Function
Sealed Spacecraft or Spacecraft with Sealed Units	1520 torr (Absolute)	100%*	Determines leak locations, not leak rate

*Desirable

2.1.2.3 Setup. Before the test, a known concentration of noncorrosive tracer gas of a type that will not damage the spacecraft shall be inserted into sealed spacecraft or sealed spacecraft components.

If there are a number of such components in the spacecraft, they may be pressurized individually with different tracer gases to aid in the location of leaks.

2.1.2.4 Selection of Gas. The type and quantity of gas shall be specified in the particular spacecraft specification. Material compatibility and electrical characteristics of the gas are considerations. In the absence of problems other than leak detection, helium should be specified and nitrogen avoided.

2.1.2.5 Leak Rate. The permissible total leak rate shall be established by the GSFC project manager, who shall consider the maintenance of pressure within the spacecraft and prevention of contamination of components for the required period of time.

2.1.2.6 Measurement. Leak rates shall be measured with a mass spectrometer. Prior to the test, the mass spectrometer shall be calibrated against a standard leak device and the magnitude of the background of the tracer gas in the test chamber shall be determined.

2.1.2.7 Leak Check. The leak check shall be performed by placing the spacecraft in a vacuum chamber which shall be reduced to 1×10^{-4} torr or less, and the chamber shall be monitored to detect leakage. The duration of the test shall depend on spacecraft characteristics and shall be stipulated in the particular spacecraft specification.

2.1.2.8 "Sniff" Test. A "sniff" test per Table II may be performed at any time during the sequence of environmental tests to detect leak locations. This test does not provide leak rate information and so must be used in conjunction with the vacuum chamber test specified above.

2.1.3 Electrical Performance

2.1.3.1 Purpose. The purpose of this test is to verify electrical performance of all systems during the spacecraft design qualification environmental test program. Satisfactory electrical performance in all applicable modes before, during, and after the specified environments shall be required prior to approval of the spacecraft design for fabrication of the flight spacecraft.

2.1.3.2 Times of Performance. The initial test shall be conducted prior to the environmental tests to determine if electrical performance meets the requirements of the particular spacecraft specification. The electrical performance test shall be repeated before, during, and after each exposure in the spin, temperature, humidity, vibration, acceleration, thermal-vacuum and solar simulation tests (as specified in the provisions for these tests) to determine if these exposures adversely affect performance. Electrical performance shall be checked for all mechanical and electrical modes of spacecraft operation during the operational and mechanical functioning test phase, 2.1.16.

2.1.3.3 Initial Test

(a) Purpose. Besides determining if performance meets the requirements of the particular spacecraft specification, the initial test establishes reference values from which to determine if succeeding electrical tests show degradation of performance. Since each spacecraft has unique electrical performance characteristics, the reference data must be established for each individual spacecraft.

(b) Levels and Measurements. The test requires application of operational loads to the spacecraft which will result in expected parameters of voltage, impedance, and current as well as expected pulse timing and waveform for the spacecraft components, subsystems, and experiments. These parameters shall be varied throughout the parameter ranges in a manner which approximates the sequence and levels expected in all normal modes of flight operations. It is desirable, though not mandatory, to vary the parameters over a broader range which simulates worst-case conditions to determine operating limits during environmental exposure. When appropriate for the mission, the spacecraft shall be stimulated by radioactive sources or other energy application.

The responses to applied electrical loads and stimuli shall be measured through radiated telemetry. The telemetry output shall be compared with hard line measurements (if available) to aid in performance evaluation.

(c) Documentation. A record shall be made of all data necessary to determine that spacecraft performance meets the requirements of the particular spacecraft specification and to provide a basis for determining satisfactory performance in the subsequent performance tests conducted before, during, and after the environmental exposures.

(d) Conditions. These tests shall be conducted under standard conditions as defined in 1.9.4.

2.1.3.4 Succeeding Tests

(a) Levels and Measurements. The operational loads and measurement methods stipulated in the initial test (2.1.3.3) shall be used in the subsequent electrical performance tests specified under design qualification except that the tests conducted during the spin, vibration, acceleration, thermal-vacuum and solar simulation exposures shall be confined to the operational modes applicable to the particular exposure.

(b) Documentation. The responsible test engineer shall maintain a chronological log of all electrical performance periods and durations. The log shall show for any particular time the spacecraft operational mode, its mechanical configuration, and the environmental exposure and stimulus being applied.

(c) Conditions. Tests not conducted during environmental exposures shall be conducted under standard conditions as defined in 1.9.4.

2.1.3.5 Instrumentation Precautions. The following precautions shall be taken in design of test instrumentation:

(1) Instrumentation shall be designed to prevent interference on any test hard lines.

(2) Any test hard lines shall be terminated in a manner to prevent damage to spacecraft by inadvertent grounding or shorting.

(3) When it is necessary to supply the spacecraft from an external power source, the source output shall be limited so that the maximum spacecraft voltage and current cannot be exceeded.

(4) All monitoring instruments shall have floating-type inputs and input impedances greater than 100K ohms to prevent abnormal drains from the spacecraft power system.

(5) Spacecraft pyrotechnic systems should be in a safe condition at all times except when the checks and test stipulated in 2.1.4 are being conducted. Inertness should be determined prior to all electrical performance tests which do not include checking pyrotechnic-spacecraft system interactions.

2.1.3.6 Test Procedures. The following techniques shall be employed in the conduct of each electrical performance test.

(a) Experiment Activation. When feasible, experiments initially shall be activated separately by appropriate stimuli to obtain reference data for isolated operation of each. Then experiments which operate simultaneously in flight shall be activated together to obtain full system reference data and expose any adverse interaction as well as electromagnetic interference.

Care shall be taken that applied stimuli do not damage the spacecraft or degrade its operation.

(b) Power Supplies. If power supplies supplant batteries during the test, the battery terminal impedance shall be simulated. If power supplies supplant solar paddles during the test, the solar cell array impedance shall be simulated.

2.1.4 Pyrotechnics Performance

2.1.4.1 Precaution. The spacecraft pyrotechnics should be in a safe (disarmed) condition at all times other than when the checks and tests stipulated below are being conducted.

2.1.4.2 Checking Circuitry. The pyrotechnic circuitry (including timers, etc.) and its interaction with the spacecraft electrical system should be checked with the use of simulators having electrical characteristics identical to those of the pyrotechnics.

2.1.4.3 When to Check Circuitry. Checks should be conducted during the initial spacecraft electrical performance test and during the electrical performance tests conducted after the temperature-humidity test phase, after the thermal-vacuum test phase, and after the vibration test phase. If the pyrotechnic circuitry is integrated sufficiently with the spacecraft to permit performance checks during vibration exposures, the pyrotechnics circuitry shall be checked during those periods also.

2.1.4.4 Operational and Deployment Tests. During this test phase (2.1.16) all pyrotechnic devices are armed so that they may experience the simulated launch and flight conditions and so that assurance may be gained that they perform their assigned functions in actuating despin, deployment of appendage, separation, etc.

2.1.5 Balance

2.1.5.1 Central Requirements. Each spacecraft shall be balanced to satisfy its orbital requirements as noted in 2.1.5.6. The launch phase also places balance requirements on the spacecraft according to the launch vehicle used (see initial balance table in the applicable launch vehicle appendix).

It is an objective of design qualification analytical balancing operations to evaluate the adequacy of spacecraft quality control in areas which could affect feasibility of attaining balance requirements for flight spacecraft.

Balancing is chosen as an early operation so that the method of attaching the balance weights to the spacecraft and the effect of the balance weights on the operation of the spacecraft may be evaluated during the course of environmental tests. The spacecraft shall be balanced while in a nonoperative state.

2.1.5.2 Correcting Unbalance. To correct unbalance, weights shall be attached, removed and/or relocated as approved by the designated representative of the GSFC project manager.

Necessary spacecraft modifications, including optimizing the location of components, shall be done with the approval of the spacecraft contractor and a designated representative of the project manager. The amount of residual unbalance for both launch and orbital configurations shall be measured and recorded for comparison with the balance requirement of the particular spacecraft specification. The spin rate used in balancing any configuration of the spacecraft shall not normally exceed that expected in flight. Balance operations shall include interface fit and alignment checks as necessary to insure alignment of geometric axes compatible with balance requirements.

2.1.5.3 Analytical Balancing. Balancing operations shall include measurement and tabulation of physical parameters (weight and mass center location referenced to spacecraft coordinates) of appendages, retro motors, and other system elements which may not be assembled for spin balancing. This data shall be processed to determine unbalance contributed by those elements for launch and orbital configuration.

The facilities and procedures for analytical balancing shall have been fully defined at the time of design qualification balance, including sufficient exploratory analytical balancing operations to provide confidence that the final flight acceptance balance can be performed satisfactorily and expeditiously.

2.1.5.4 Measurement Techniques. Measurement techniques include compensation and/or correction for the effects of yield, assembly tolerances, spin, and gravity (2.1.16.4b) on the accuracy and repeatability of the measurements. Unbalance attributable to imperfect control of these factors should not exceed 50% of the final flight acceptance balance specified in the applicable launch vehicle appendix.

2.1.5.5 Launch Configuration. The current design restraints manual for the launch vehicle forms the basis for the launch configuration balance requirements. Requirements for the prototype spacecraft allow twice the unbalance permitted in the final balance of the flight spacecraft and are stated in Table 1 of the applicable launch vehicle appendix.

2.1.5.6 Orbital Configuration. Orbital balance requirements, based on the particular spacecraft mission, shall be furnished by the project manager. Balance requirements and procedures shall appear in the particular spacecraft specification.

2.1.6 <u>Spin</u>

2.1.6.1 Applicability. This test applies to spin-stabilized spacecraft only. It is intended that it be performed after balancing while the spacecraft is still on the balancing machine. A spin test at this time provides an early opportunity to check spacecraft electrical performance per 2.1.3 while the spacecraft is exposed to steady-state loads.

This simple test may be waived when, as is usually the case, the spacecraft is subjected to more severe steady state and dynamic loads during the operational and mechanical functioning spin tests of 2.1.16.

2.1.6.2 Test Conditions. The spacecraft structure shall be visually examined for material yield before and after exposure to the spin. Before, during and after the test, all applicable electrical systems shall be turned on and monitored, except for high voltage systems which would be inappropriately subject to corona due to the incorrect atmospheric pressure.

2.1.6.3 Spin Rate. As shown in Table III, spin speed shall be 1-1/4 times the nominal speed during launch or orbit, whichever is greater. Spin shall be maintained for 10 minutes, longer if necessary to verify spacecraft operations.

TABLE III

SPIN
SPACECRAFT DESIGN QUALIFICATION

Electrical Operation	Spin Rate	Duration (min)
All Applicable Systems (2.1.6.2)	1-1/4 x nominal launch or orbital rate (whichever is greater)	10*

*Longer if necessary to verify spacecraft operations.

2.1.7 Weight, Center-of-Gravity, and Moments of Inertia

2.1.7.1 General Requirements. The parameters of weight, center-of-gravity, and moments of inertia are used in predicting vehicle performance during launch, and spacecraft orientation during injection and orbit. The center-of-gravity and moments of inertia shall be determined for the spacecraft configuration to be employed during final stage burning and, when different from the final stage burning configuration, for the orbital flight configuration. Parameters for intermediate configurations shall be determined only as requested by the project management.

2.1.7.2 Procedures. While nonoperative, the spacecraft weight, center-of-gravity, and moments of inertia (about the spin axis and the maximum and minimum moments about the transverse axes) shall be determined and compared with design requirements.

2.1.7.3 Tolerances. Measurement tolerances shall be as specified in Table IV. It is acceptable to base calculations of the entire spacecraft's parameters on component data as long as specified accuracy can be met.

TABLE IV

PHYSICAL MEASUREMENT TOLERANCES
SPACECRAFT DESIGN QUALIFICATION

Weight	Center of Gravity		Moments of Inertia*				
			Thrust	Lateral			
	Longitudinal	Lateral**	Z-Z Axis	Max	Min	X-X Axis	Y-Y Axis
±0.25%	±0.0625 in.	±0.0625 in.	±1.5%	±1.5%	±1.5%	±1.5%	±1.5%

*Measurements shall determine inertias about principal axes as well as those about X-X, Y-Y, and Z-Z axes.

**Not required for spin stabilized spacecraft (dynamic balancing as specified in Table 1 of the applicable launch vehicle appendix gives more accurate results).

2.1.8 Temperature and Humidity

2.1.8.1 Purpose. This test phase consists of hot and cold operational tests, hot and cold storage tests, and a humidity test. The storage and humidity tests demonstrate the ability of the design to withstand the environment which might be encountered in shipment and storage of a spacecraft if no attempt were made to control ambient conditions.

The operational tests of the spacecraft are conducted before thermal-vacuum testing of the prototype is attempted. They thus serve to indicate the resistance of the design to extremes of expected in-flight temperatures, plus a safety factor, and to give some assurance that it is worthwhile to conduct the more complicated and expensive thermal-vacuum test exposure. It should be recognized that components with heavy power dissipation will not reach maximum temperatures when operated at atmospheric pressure.

2.1.8.2 Precautions. During the low temperature exposure, utmost care shall be exercised that no condensation of moisture occurs on the spacecraft or any of its components. Care also shall be taken that the rate of change of temperature does not exceed the limits of the spacecraft's thermal characteristics. The spacecraft shall not be exposed to direct air flow from the chamber fans.

2.1.8.3 Protection of Spacecraft. In general it is desirable that ground service equipment be provided to protect each spacecraft (and critical components which are handled separately) from extremes of temperature and humidity during storage, shipment, and preparations for launch. When such equipment is provided, the storage and humidity tests may be eliminated or modified accordingly. If ground service equipment is to be employed, assurance shall be gained at the time of design qualification testing that the equipment does in fact provide the protection desired. Such assurance shall be based on successful test results or on a documented, independent engineering study.

2.1.8.4 Test Setup. The spacecraft shall be supported at locations specified by thermal analyses and structural consideration. The materials utilized to support the spacecraft shall minimally influence the thermal distribution. If this is not possible, the mounting fixture shall be designed to minimize interference to radiation paths from radiant sources to the spacecraft.

The spacecraft shall be installed in the chamber in such a manner that is is not exposed to any abnormally hot or cold sources (other than those specifically intended during the test) or so that the effects of such sources are minimized. Provisions shall be made to assure that the limits of environmental temperatures specified by this document are not exceeded. Temperature shall be monitored by appropriate temperature sensors located on and in the spacecraft at positions determined from a thermal analysis. Those locations shall be noted in the the test plan for the particular spacecraft. Special thermal controls shall be provided as required to maintain spacecraft operating temperatures within safe limits.

2.1.8.5 Storage Temperature Tests. These tests shall be conducted at the levels specified in Table V, below, with the spacecraft electrical system turned off.

(a) Waiver. These tests need not be performed if environmental protection is provided per 2.1.8.3, or if the temperature extremes stipulated by the thermal-vacuum and thermal balance test phase (2.1.13) are equal to or exceed the levels in Table V.

(b) Electrical performance. Spacecraft electrical performance shall be checked per 2.1.3 after the test setup for each exposure has been completed and the chamber closed. At the conclusion of each exposure, the spacecraft performance test shall be repeated at standard conditions to detect any adverse effects from the preceding exposure.

TABLE V

STORAGE TEMPERATURE TEST SCHEDULE
SPACECRAFT DESIGN QUALIFICATION

Cold Exposure		Hot Exposure	
Stabliized Temperature	Duration	Stabilized Temperature	Duration
-30°C	6 hours	60°C	6 hours

2.1.8.6 Operational Temperature Tests. These tests shall be conducted at the levels specified in Table VI. Electrical performance tests per 2.1.3 shall be conducted prior to and after each soak. Electrical performance testing during the soaks shall be performed as stated below and in 2.1.3.

Before the chamber temperature is lowered for the cold soak, the chamber dehumidification system shall be operated to reduce the moisture content of the chamber air to a minimum. Chamber dehumidification shall be continued during cool-down until 0°C is reached. During warm-up periods, all spacecraft temperatures shall be maintained above the dew point temperature of the chamber air.

The soak test temperatures specified in Table VI shall be approached in no greater than 10°C increments. At each increment all on-board sensors shall be checked for calibration by the spacecraft telemetry. Stabilization at the incremental levels is not required. During transition to the cold soak temperature, operation of the spacecraft shall be limited to the extent necessary for the calibration checks obtained by the spacecraft telemetry.

Upon reaching the soak test limits, the spacecraft shall be operated for a time sufficient to produce steady-state temperature operation of the heat-producing subsystems of the spacecraft. The spacecraft operation shall be checked during this period.

When temperature gradients greater than 20°C are predicted or expected in the spacecraft, provisions shall be made when feasible for local heating or cooling as required to simulate realistic gradients.

When applicable for the mission, spacecraft cold start capability shall be demonstrated at least three times. Each cycle of operation shall start by reverting to the stabilized cold condition specified in Table VI.

TABLE VI

OPERATIONAL TEMPERATURE TEST SCHEDULE
SPACECRAFT DESIGN QUALIFICATION

Cold Exposure*		Hot Exposure	
Orbital Temperature	Operational Duration	Power-off Orbital Temperature	Operational Duration
10°C below predicted	Until temperature stabilized +6 hours	10°C above predicted	Until temperature stabilized +6 hours

*For spacecraft having undervoltage-recycle capability or other devices which deactivate heat-producing components, subsystems, and experiments, the cold soak temperature shall be 10°C below the cold extreme predicted for the mission with the spacecraft in a "power-off" status. For those spacecraft which do not have a planned deactivation mode, the cold soak temperature shall be 10°C below the minimum component/subsystem/experiment temperature predicted for the mission.

2.1.8.7 Humidity. Spacecraft shall be subjected to a humidity test to the levels specified in Table VII, below, unless environmental protection is provided by ground support equipment per 2.1.8.3. The test shall be performed in a chamber in which the air is constantly circulated during the test.

The temperature of the spacecraft shall be maintained throughout the test above the dew point temperature to preclude condensation. Applicability of this requirement shall be stated in the particular spacecraft specification.

Electrical performance tests shall be conducted before and after the humidity test per 2.1.3. The humidity test shall be conducted with the spacecraft turned off. At the completion of the twenty-four hour exposure period the chamber shall be dehumidified to standard conditions and the spacecraft operated to check for any deleterious effects arising from the exposure.

TABLE VII

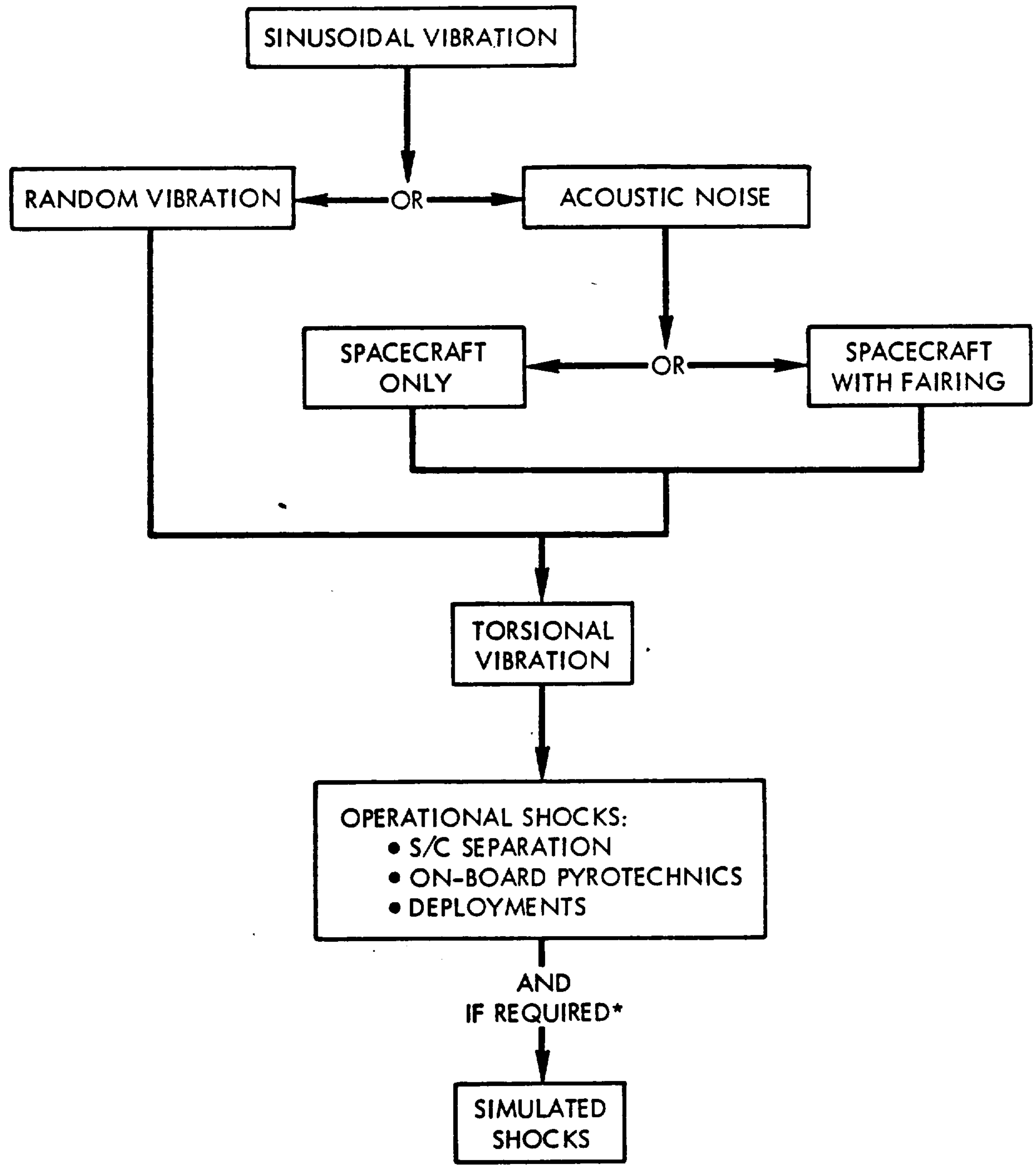
HUMIDITY TEST SCHEDULE
SPACECRAFT DESIGN QUALIFICATION

Stabilized Temperature	Relative Humidity	Exposure
30°C	95% +0 -5% rh	24 hours

2.1.9 <u>Vibration</u>

2.1.9.1 Relationship to Other Structural Dynamic Tests

(a) Single Environment Tests. Shock, sinusoidal and random vibration, and acoustic noise tests are interrelated and in some cases interchangeable. Figure I shows the various paths which may be used to form an acceptable qualification test program. The



*See paragraph 2.1.11.2.

Figure 1–Interrelationship of Structural Dynamic Tests.

limitations of simulating vibration should be recognized in establishing the program to be followed for a given spacecraft. In particular, random excitation of subassemblies arises both from airborne and structure-borne inputs. Because of the difficulty of achieving proper simulation of the entire shroud-vehicle-spacecraft-mounting configuration, it is possible that an acoustic test would not excite subassemblies near the mounting interface to a sufficient degree. Similarly, because of the transmission characteristics of the structure, the random excitation applied through the interface is unlikely to provide sufficient excitation at the forward end of the spacecraft. Because of these deficiencies, the exposure of subassemblies to random vibration on an individual basis is considered to be particularly important.

An acoustic noise test generally is preferred to a random vibration test for larger spacecraft, which are apt to be more sensitive to the acoustic environment. Smaller spacecraft, which are more likely to respond to directly applied vibration, shall be subjected to random vibration testing.

(b) Combined Environment Tests. Dynamic forces generated during launch may combine and interact in such a way as to damage a spacecraft capable of withstanding the same forces applied singly. If analysis indicates significant interaction, the prototype spacecraft should, insofar as practicable, be subjected to the combined environments. The launch environments which are expected to be the most significant when combined are acceleration, acoustic noise, vibration and pressure profile. To be meaningful, the time variation of the environments must be considered.

2.1.9.2 Proto-Flight Spacecraft. Proto-flight spacecraft, as defined in 4.11, are tested at design qualification levels but at flight acceptance durations; that is, at one-half the durations called for in the referenced design qualification tables.

2.1.9.3 Retro and Apogee Motor Vibration. In addition to vibration tests associated with the launch phase, spacecraft equipped with retro or apogee motors shall undergo a test simulating the vibration that occurs in that type of final boost. Such tests shall be conducted with the spacecraft attached to the vibration source at the apogee/retro motor adapter. The spacecraft shall be in the apogee/retro motor burning mode of operation during the test and, when applicable, with booms and paddles extended (and supported as required to simulate flight gravity conditions). Levels and procedures for this test shall be stated in the particular spacecraft specification.

2.1.9.4 Spacecraft Performance. Before and after each vibration exposure, the spacecraft shall be examined and functionally tested to check performance. During the vibration test, the spacecraft shall be operated in a duty cycle typical of that to be employed in the launch phase and monitored for malfunctions in telemetry and all other systems which operate during boost. Exact requirements for such monitoring shall be specified in the test plan for the particular spacecraft.

2.1.9.5 Spacecraft Setup. The spacecraft shall be attached to a vibration fixture by use of a flight-type spacecraft adapter and flight-type clamp. Vibration shall be applied at the base of the adapter via the fixture in each of three orthogonal directions, one direction being parallel with the thrust axis. Antennas and other devices which extend or change position after orbital injection shall be in the launch configuration during the test.

2.1.9.6 Internal Pressure Considerations. Normally sealed components shall be pressurized during test to their prelaunch pressure. In cases where significant changes in strength stiffness, or applied loads result from variations in internal or external pressure during the launch phase, a special vibration test should be considered to cover those effects.

2.1.9.7 Test Fixtures and Support Tables

(a) Design Goals. The design and fabrication of ideal test fixtures and support tables for the larger interfaces are beyond present state of the art. Realistic design goals in the development of the test system, however, should include the following minimum requirements:

(1) The horizontal support table should be extremely rigid so as to allow virtually no rigid body motion except in the direction of applied vibration.

(2) The lowest resonant frequency of the test fixture should be well above the major structural resonances of the spacecraft/adapter combination. A resonance not lower than 1000 Hz is a reasonable requirement for a fixture designed for an 18-inch diameter adapter. For fixtures designed for larger adapters, successively lower resonant frequencies are acceptable. For example, the fixture for an adapter of 60-inch diameter may have a resonance of 250 Hz.

(3) Material and structural damping of the fixture should be maximized when possible to minimize the effects of resonances.

(b) Fixture Survey. A resonance survey shall be conducted on the test fixtures and support tables prior to any spacecraft tests. The results of this survey provide a check on the dynamic characteristics of the fixtures and serve as a basis for selecting the best points for controlling vibration input and for locating the recording accelerometer.

2.1.9.8 Vibration Control Instrumentation. For the purpose of controlling vibration applied to the spacecraft, calibrated accelerometer(s) shall be attached rigidly on the test fixture near the fixture-spacecraft adapter interface and aligned with the axis of applied vibration.

The overall control accelerometer signal system shall be calibrated for frequency response from 5 Hz to the upper frequency of the test and for amplitude linearity characteristics to values 1.5 times the maximum input signals expected to be applied during the tests.

Accelerometers used for control shall be selected from those types known to have a low base strain sensitivity as defined in 4.19.

(a) Multi-Point Control of Large Fixtures. Because of the resonance characteristics of the large test fixtures, no single point is apt to be representative of the vibration input across the interface. Moreover, control from a single point is likely to be difficult because of the wide range of dynamic response resulting from resonances. A practical means for circumventing these problems is to average the signal levels from a group of control accelerometers distributed around the adapter base. Results of the test fixture survey will aid in selecting the number and location of the accelerometers. Usually, four such accelerometers are adequate.

For sinusoidal tests, the "average" should be the arithmetic mean of the peak signal amplitudes from the control accelerometers and not the average of the instantaneous values. Thus, the averaging device should be insensitive to the phase of the signals.

For random vibration tests, a device which effectively averages the power spectral densities at each frequency may be used.

(b) Single-Point Control. If averaging is not practicable, the results of the test fixture vibration survey shall be used to determine the best location for single-point control with one control accelerometer.

2.1.9.9 Vibration Recording Instrumentation. For recording the applied vibration and responses, calibrated accelerometers shall be positioned and rigidly attached adjacent to the control accelerometer(s) at the fixture/adapter interface and on the spacecraft. It is preferred that the control accelerometer(s) be utilized as control monitor accelerometer(s) if there is a laboratory capability for simultaneously controlling and recording. In addition, two control monitor accelerometers to measure cross axis motion shall be mounted, each perpendicular to the control accelerometer(s), and aligned with the remaining two axes (X-X, Y-Y, or Z-Z axis) as applicable. Other vibration sensing devices shall be attached to the structure and subsystems in critical locations as dictated by component specification requirements, dynamic analysis, etc. They shall be used to define the overall vibration characteristics without being unnecessarily influenced by local responses. Magnetic tape shall be used for recording the control or control monitor signal and, when possible, the response signals.

2.1.9.10 Sinusoidal Vibration

(a) Levels. This portion of the test should be conducted by sweeping the applied vibration once through each frequency range specified in Table 2 of the applicable launch vehicle appendix.

(b) Filtering. During sinusoidal testing, the waveform of the input is likely to be distorted because of non-linearities within the spacecraft. Such distortions can have serious effect on the vibrator's control system causing a possible overtest or undertest. It is therefore necessary, in order to meet the tolerances of this specification on amplitude, i.e. ±10%, to filter the signal from the control accelerometer before it enters the automatic control circuit of the vibrator. A tracking bandpass filter is required to remove distortion from the control signal. The bandwidth of the filter shall be 10 Hz or less for frequencies below 80 Hz, and 100 Hz or less for frequencies between 80 and 2000 Hz. The center frequency of the filter shall be capable of following the oscillator driving frequency within 20% of the filter bandwidth.

The use of a tracking bandpass filter for controlling input acceleration in the resonance region may result, in some cases, in very low control accelerometer signal levels which approach the noise level of the shaker system. In that event, the notch level may be controlled by using the unfiltered control accelerometer signal. However, the use of this approach requires that the notch level be established from low-level surveys using the unfiltered control accelerometer signal as a reference.

(c) Resonance Considerations

1. Different Spacecraft Characteristics—It is recognized that during launch the dynamic characteristics of a spacecraft influence its own vibration environment. The influence is characterized, in general, by a reduction in input levels to the spacecraft at the primary spacecraft resonant frequencies. That is, the dynamic response of the large spacecraft will tend to suppress the vehicle response and thus significantly affect the motion of the base of the spacecraft. The levels specified in the appendix table do not take into account the possible reduction at resonance because the effect is a function of the dynamic characteristics of each spacecraft. Also, it is not within the state of the art to exactly duplicate in the test laboratory the boundary conditions and mechanical impedance that the spacecraft "sees" when it is attached to the launch vehicle. To preclude failure from unrealistic loads, spacecraft weighing more than 200 lbs. should be critically reviewed prior to sinusoidal vibration testing.

2. Reducing Levels or "Notching" (see 4.20)—Where it can be shown by accepted dynamic analysis techniques that a particular spacecraft in combination with the launch vehicle experiences loads at all critical flight conditions less severe than those induced by the specified vibration test, the test levels may be reduced for the spacecraft/adapter combination in the frequency bands of resonances of primary structure. In such analysis, the comparison of calculated flight levels and flight acceptance test levels is valid only where the assumed forcing functions acting on the vehicle/spacecraft analytical model are taken at the mean plus 2 sigma or 97.7% probability level for each critical flight condition. The comparisons shall be made at selected critical load locations on primary structure based on the combined member stresses at such locations. The GSFC spacecraft project manager shall coordinate the combined vehicle/spacecraft loads analysis with the launch vehicle manager.

3. Preferred Methods of Notching—Once it has been established that the test inputs may be limited at resonance, one of several methods may be employed. The most direct is to monitor strain at critical load locations. Indirect methods may be used in cases where the functional relationship between modal response and critical member stress levels has been determined. Accelerometers or other motion transducers attached to the structure may be used for monitoring modal response. If critical member stress levels have been determined as a function of input vibratory acceleration and frequency, the third and simplest method, from an operational standpoint, is to limit the induced stress by introducing a narrow-band notch into the input acceleration program.

   Either manual or automatic control methods may be used in programming notch width and depth. In the automatic system the control parameter is automatically switched from "input acceleration" to "allowable response" and back as the sweep progresses. It is the more desirable method since the full allowable stress value is maintained across the entire notch.

   The manual method involves use of a predetermined notch, usually of rectangular shape. The bandwidth of the notch should be determined immediately before the test by a low-level sine sweep survey, and <u>not</u> by extrapolated values from previous tests of the same or similar spacecraft. The survey level should be as high as possible to minimize nonlinear effects. The notch bandwidth must be narrow enough to develop 90% of the allowable loads at the band edges.

4. Alternative Method of Notching—As an alternative to the preferred methods above, the response may be limited at resonances so that the design strength of the structure will not be exceeded. For this method to be acceptable, the GSFC spacecraft project manager shall establish that the load factors used in the spacecraft structural design are conservative enough to encompass all critical

flight conditions, including combined static and dynamic loads. In that case, levels should be limited by any of the techniques suggested in the previous paragraphs. The member stresses corresponding to the design load factors shall be determined by stress analysis and should, if possible, be verified by static load or acceleration tests.

5. Application to Flight Acceptance Test—The aforementioned procedures are applicable to flight acceptance as well as design qualification testing; the only difference being that the flight acceptance levels are based on the 2 sigma flight loads and the design qualification levels are 1.5 times those. Sweep rates and other parameters will be in accordance with Table 2 in the applicable launch vehicle appendix.

2.1.9.11 Torsional Vibration. Because certain launch vehicles impart significant torsional oscillations to the spacecraft, launch vehicle flight data must be reviewed and, in some cases, analytical results must be obtained to assess the structural adequacy of the spacecraft and its components.

In such cases where torsional responses of the spacecraft are not clearly enveloped by translational vibration and/or static loads tests, a torsional vibration test of the prototype spacecraft is a requirement. Instrumentation, fixtures, tolerances, etc., shall correspond to those specified for translational vibration.

2.1.9.12 Random Vibration

(a) Levels. Gaussian random vibration shall be applied as specified in the random vibration table of the applicable launch vehicle appendix.

(b) Control. With the spacecraft and adapter mounted on the vibrator, the excitation spectrum, measured from the control accelerometer(s), shall be equalized such that the power spectral density is within ±3 db of the specified levels everywhere in the frequency band and the overall RMS level is within ±10% of that specified. The spectrum analyzer must have the following characteristics:

1. Real time, parallel filter analyzers:

   a. Filter bandwidths shall not exceed 25 Hz below 1200 Hz or 100 Hz above 1200 Hz.

   b. Averaging times shall be at least 2.5 seconds for each filter band where the bandwidth is 10 Hz or greater. For narrower filter bandwidths, the averaging time shall be at least 25 divided by the filter bandwidth.

2. Tape loop analyzers:

   a. Filter bandwidth must be equal to or less than those specified in paragraph 1.a, above.

   b. The record length (seconds) shall be at least 150 divided by the filter bandwidth.

   c. The averaging time of the analyzer should be approximately equal to the record lengths.

   d. The frequency scan rate (Hz/second) shall be less than the filter bandwidth divided by the averaging time where true integrating is used, or filter bandwidth divided by four times the averaging time where resistance-capacitance integrating is used.

2.1.10 Acoustic Noise

2.1.10.1 Relationship to Other Structural Dynamic Tests. For the relationship of the acoustic noise test to other structural dynamic tests see 2.1.9.1. If random vibration tests are conducted, the acoustic noise test generally is not required. See 2.1.9.2 and 4.11 for proto-flight spacecraft test levels and durations.

2.1.10.2 Spacecraft Performance. Prior to and after the acoustic noise exposure, the spacecraft shall be examined and functionally tested to check performance. During the test the spacecraft shall be operated in a duty cycle typical of that to be employed in the launch phase and monitored for malfunctions in telemetry and all other systems which operate during boost. Exact requirements for such monitoring shall be specified in the test plan for the particular spacecraft test program.

2.1.10.3 Internal Pressure Considerations. Normally sealed components shall be pressurized during test to their pre-launch pressure. If significant changes in strength, stiffness, or applied loads result from variations of internal or external pressure during the launch phase, a special acoustic test should be considered to cover those effects.

2.1.10.4 Methods of Testing. The spacecraft shall be subjected to the acoustic noise specified in Table 4 of the applicable launch vehicle appendix. Two methods of testing, with or without a flight-type shroud, are acceptable.

(a) With Shroud. The spacecraft shall be mounted on a flight-type adapter. The adapter, in turn, is mounted to a structure that, if practicable, is dynamically similar to the vehicle structure immediately aft of the spacecraft adapter. The assembly with the shroud shall be placed in a progressive wave chamber so that the direction of propagation is parallel to the longitudinal axis of the assembly. A minimum of three microphones at each of two longitudinal stations, one near the top of the shroud and one near the bottom, shall be used for control and verification of the acoustic spectrum.

(b) Without Shroud. The spacecraft shall be mounted on a flight-type adapter and tested in a reverberant noise chamber to the levels of Table 4 of the applicable launch vehicle appendix. A minimum of three microphones at each of two spacecraft stations, near either end, shall be used for control and verification of the acoustic spectrum.

2.1.10.5 Tolerances. The measured overall levels during test shall be within +3 and -1 db of the specified value. The octave band levels shall be ±3 db of those specified except that the bands with center frequencies of 15.8 Hz, 31.5 Hz, 4000 Hz, and 8,000 Hz shall be within ±5 db of those specified.

2.1.11 Shock

2.1.11.1 Relationship to Other Tests. For the relationship of the shock test to vibration and acoustic noise tests see 2.1.9.1. Shock requirements are also related to the separation and deployment tests which are performed under 2.1.16 in that those tests may sometimes serve a dual role in satisfying the shock requirements prescribed below. See 2.1.9.2 and 4.11 for proto-flight spacecraft test levels and durations.

2.1.11.2 Vehicle-Induced Shocks. Shocks due to events such as engine ignition and burnout, stage separation, and shroud ejection must be considered in the spacecraft qualification program. An envelope of the launch vehicle shocks induced at the spacecraft/launch vehicle interface is presented in Figure 1 of the applicable launch vehicle appendix.

A pulse or complex transient whose positive and negative shock spectrum matches that defined in the appendix within the tolerances of +50% and -10% shall be applied to the spacecraft/launch vehicle interface twice along each of the three major axes.

Any one of a variety of shock simulators including electrodynamic shakers, fluid powered actuators, drop testers and devices powered by pyrotechnic charges may be used.

It must be emphasized that the critical damping ratio (c/c_c) used in the shock spectral analysis of the test pulse must equal the damping ratio specified in Figure 1 in the appendix. This ratio is usually $c/c_c = .05$, which corresponds to a single-degree-of-freedom system with a maximum amplification Q of 10.

2.1.11.3 Spacecraft Separation Shock. The separation of the spacecraft from the final stage of the launch vehicle and adapter is usually accomplished by disengaging a clamp using a pyrotechnic device. The resultant shock at the separation plane is normally of very high level (1000 g typically) and of high frequency content.

Two spacecraft separation tests are required for the prototype spacecraft.

Besides the spacecraft, the test will include a flight-type clamp and pyrotechnics, the flight-type adapter, and fixtures and suspension systems which will allow separation to occur.

The usual technique is to suspend the connected spacecraft and adapter by the spacecraft and allow the adapter to fall after separation. Since flight-type hardware is used, this test does not contain the usual prototype test factor of 1.5 above the anticipated actual separation shock.

The separation test may, in some instances, produce shocks which are more severe than the launch phase shock levels (2.1.11.2). Where appropriate instrumentation demonstrates this, the launch phase shock test need not be performed.

2.1.11.4 Post Launch Shocks. Shock excitation will occur after launch and separation when pyrotechnic and pneumatic devices are actuated to release booms, paddles, and protective covers. Also, the impact of deployable devices at the end of their stroke is a likely source of significant shock.

These shocks, and in some cases the separation shock (2.1.11.3), may be performed as part of the Operational and Mechanical Functioning tests (2.1.16).

The tests shall be performed twice on the operating prototype spacecraft.

2.1.11.5 Retro and Apogee Motors. For spacecraft with this type of final propulsion, shocks from motor ignition, burning and burnout must be considered. If these shocks are more severe than those already provided for, a shock test covering the motor characteristics shall be included in the particular spacecraft specification

2.1.11.6 Transportation and Handling. Environmental conditions resulting from transportation and handling could exceed the levels specified herein, which are based on the launch environment. However, it is felt that spacecraft design should not be penalized by requiring that unprotected spacecraft withstand these conditions. Instead, the shipping and handling environment should be modified by controlling the modes of transportation and handling methods, and by use of properly designed shipping containers.

Assurance that adequate protective devices and procedures are to be employed during handling and transportation shall be provided by documented analysis and test results.

2.1.12 Structural Loads and Acceleration, Steady State

2.1.12.1 General. The ability of the spacecraft and its subsystems to withstand loads imposed by combined steady state and dynamic acceleration shall be demonstrated by one of the following:

(1) an acceleration test on the complete prototype* spacecraft

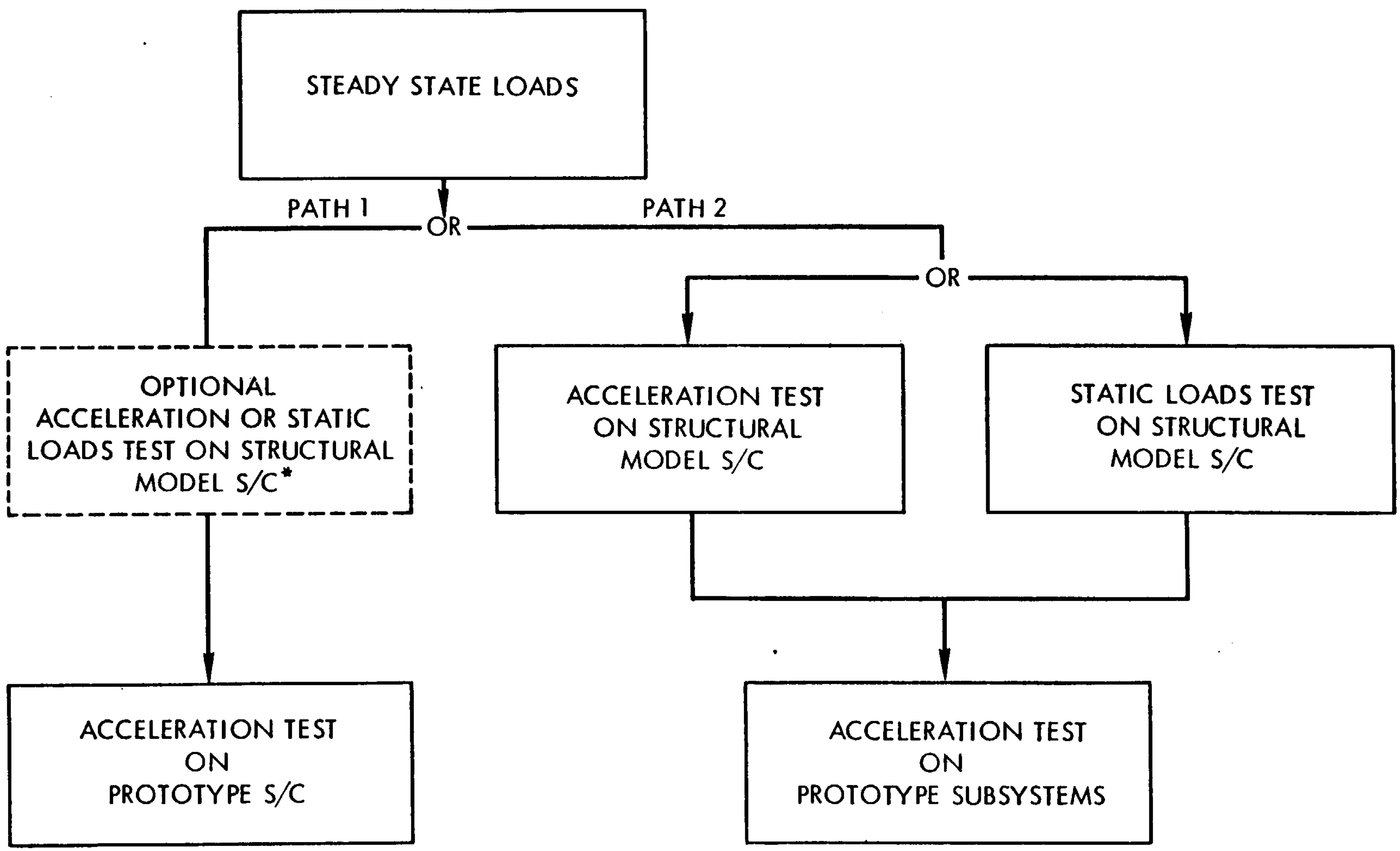
(2) an acceleration test on the structural model spacecraft with a separate acceleration test on the prototype subsystems

(3) static loads test on the structural model spacecraft and an acceleration test on the subsystems.

2.1.12.2 The Alternative Paths of Qualification. Figure II illustrates the paths which may be used to conduct a satisfactory design qualification test program. If facilities are adequate, the most straightforward approach, in most cases, is an acceleration test on the prototype spacecraft as shown in path 1 of Figure II. In that approach, as indicated by the broken-line box, an acceleration or static loads test may first be performed on the structural model to gain assurance that the prototype spacecraft would not experience a catastrophic failure when it is tested.

In path 2, either an acceleration test or a static loads test is performed on the structural model spacecraft. If that approach is chosen, an acceleration test must be run on the prototype subsystems to complete the qualification requirements.

*For purposes of this section, where separate prototype and flight spacecraft do not exist, the single proto-flight unit is considered a prototype.



*May be run for confidence that the prototype spacecraft would not experience catastrophic failure.

Figure 11–Alternative Paths for Applying the Structural Loads and Acceleration Tests.

2.1.12.3 Choosing Between Acceleration and Static Loads Tests. The following points should be among those considered when choosing between the acceleration and static loads tests:

(1) Which test most closely approximates the flight-imposed load distribution?

(2) Which can be applied with the greatest accuracy?

(3) Which best provides the essential, verifying design engineering information, such as that required for predicting design capability for future spacecraft or launch vehicle modifications?

(4) Which poses the least risk to the spacecraft in terms of handling and test hardware?

(5) Which best meets cost, time, and facility limitations?

In either test, the levels applied must establish a margin of confidence which encompasses all the uncertainties associated with the test, and a structural loads and stress analysis should be performed before testing.

2.1.12.4 Acceleration Levels. Test levels are given in the structural loads table and figure of the applicable launch vehicle appendix. It is intended that these levels will demonstrate the capability of the spacecraft to withstand the most severe combination of sustained and low frequency acceleration in flight. To achieve this, the test accelerations are higher than actual measured steady accelerations, particularly in the lateral axis. Although experience has shown that this procedure demonstrates the adequacy of the spacecraft structure, the final judgment on such adequacy must be based on dynamic considerations from a combined spacecraft-launch vehicle dynamics analysis.

Where a combined thrust and lateral load is specified, at least two acceleration tests are required so that the lateral component of acceleration is applied in two perpendicular axes.

The prescribed levels for all centrifuge tests shall be applied to the base of the spacecraft adapter subject to the tolerance stated in 2.1.12.8.

2.1.12.5 Retro and Apogee Motors. For spacecraft with this type of final boost, an additional test shall be required in any of the following cases:

(1) The acceleration is in the opposite direction to the launch phase acceleration.

(2) The acceleration-induced loads are transmitted by other structure in the spacecraft.

(3) The acceleration is in the same direction as the launch phase acceleration and the acceleration of this final boost exceeds the launch phase acceleration by 10%.

2.1.12.6 Spacecraft Performance. Prior to and after acceleration test, the spacecraft shall be visually examined and functionally tested to check performance. During the tests, the spacecraft performance shall be monitored as specified in the test plan.

2.1.12.7 Test Conditions. Acceleration tests shall be conducted with sealed units pressurized to 760 torr (15 psi) in excess of prelaunch pressure in cases where changes in pressure may result in significant changes in strength, stiffness or leak properties. The spacecraft shall be in an operational state normal for powered flight.

2.1.12.8 Tolerances. The acceleration level specified in the appendix shall be held to within 0 to -5% at the bottom of the adapter. The centrifuge should be large enough to prevent the acceleration level at the extreme top of the spacecraft from being less than 75% of the acceleration level at the bottom of the adapter.

2.1.12.9 Test Setup. The spacecraft shall be attached to the centrifuge with a fixture designed to tilt the spacecraft and adapter at the proper angle for obtaining the acceleration loadings specified in the appendix. To determine the proper angle, the effect of gravity should be taken into account in each case.

2.1.13 Thermal Balance and Thermal-Vacuum Performance

2.1.13.1 Purpose and General Requirements. The thermal balance test has the purpose of evaluating the performance of the spacecraft thermal control system under specific energy boundary conditions that are related to the orbital mission. The thermal-vacuum test evaluates the performance of the other spacecraft systems under predicted vacuum but with temperature extremes 10°C more severe than predicted.

2.1.13.2 Combined Tests. It is possible that the two tests may be combined when the spacecraft design is such that the available space simulator can accurately duplicate the orbital external thermal environment. In that case the thermal balance test can be conducted with only minor dependence on analytic thermal model predictions. If the orbital boundary flux and the internal heat dissipation resulting from spacecraft operating modes are duplicated, the worst-case boundary conditions and attendant extreme nodal temperatures will result. In such cases, the practicability of performing a combined thermal balance and thermal-vacuum test is obvious and is most often technically and economically advantageous. The duration of the combined thermal balance and thermal-vacuum performance tests shall be sufficient to meet the requirements of 2.1.13.4 and 2.1.13.5.

The test plan shall indicate whether combined or separate tests are to be conducted. The order and procedures for all testing shall be stated in the particular spacecraft specification and approved by the GSFC project manager. The provisions of 2.1.3 for electrical performance testing shall apply to both tests.

2.1.13.3 Special Tests. Special tests may be required to evaluate unique features such as an attitude control system. Also, special temperature-vacuum tests may be necessary to evaluate the performance of external devices such as solar array hinges, antennae, and experiment booms which must operate soon after injection into orbit. The test conditions shall reflect, as nearly as practicable, the conditions expected in flight operations. Such special tests shall be specified in the particular spacecraft specifications under the operational and deployment test phase and shall be approved by the GSFC project manager.

During all of above tests, care shall be taken to prevent unrealistic environmental conditions.

2.1.13.4 Thermal Balance. This test shall be performed to evaluate the ability of the thermal control system to maintain the spacecraft thermal environment within established structural, experiment, and subsystem temperature limits. The test shall be conducted with the spacecraft under vacuum of at least 10^{-5} torr and thermal conditioning suitable for evaluating the particular design under consideration. The test may be performed on a thermal model, prototype, or flight spacecraft. When the test is performed on other than the flight system, the flight spacecraft acceptance test shall demonstrate duplication of the proven thermal design. This can be demonstrated in the flight acceptance program by duplicating a significant energy balance phase used in the thermal balance test. The method selected shall be stated in the particular spacecraft specification and approved by the GSFC project manager.

(a) Predicting Thermal Design Performance. For the majority of spacecraft designs and simulator capabilities it will be necessary to theoretically model the spacecraft design and orbital mission thermal environment in order to predict performance of the thermal design for the mission. These models are also used to predict the performance of the thermal design in a known chamber environment. Correlation of results between chamber thermal balance tests and the theoretical model thus provides a demonstration and means for validating the thermal design and for improving model accuracy in predicting orbital performance.

The analytical models shall be iterated to determine the conditions of maximum and minimum energy absorbed by the spacecraft from external radiant sources, direct and reflected. Significant secondary reflections and infrared emissions between external spacecraft parts are also included. Maximum, minimum, and nominal operating modes of the spacecraft shall be used in the analysis to predict total and local thermal dissipation by the internal subsystems. An appropriate (for the mission) parametric grouping and analysis of these quantities shall identify the energy combinations and nodal temperature extremes most critical to the thermal control system.

The analysis shall be performed using values of input flux, α, ϵ, and internal power dissipation so there is only one chance in one-hundred that the maximum and minimum theoretical energy balance would be exceeded during the prescribed mission. The concept of degradation of thermal control surface properties due to orbital exposure is implicit in the values selected for α and ϵ.

The approach to the thermal balance test shall be based on simulating the thermal conditions predicted by the analysis to be most critical for the thermal control system. In this context, simulation of integrated orbital or orbital dynamic energy conditions may be selected depending on the thermal time constant of the spacecraft/subsystems. Normally, the following test conditions will bound the conditions necessary for evaluating thermal design: (1) maximum external absorbed flux plus maximum internal power dissipation, (2) minimum external absorbed flux plus minimum internal power dissipation, and (3) nominal flux and dissipation.

Occasionally, because of extreme thermal isolation within the spacecraft or severe self-shadowing, additional thermal balance conditions will be required to evaluate critical features of the thermal design. It is recognized that because of the nature of the thermal design (e.g. active control, insulated designs, etc.) this approach may or may not subject all spacecraft systems to their maximum temperature extremes. Exposure of the other spacecraft systems to maximum temperature extremes must be accomplished during the thermal-vacuum performance test.

(b) Selection of Simulation Method. The external boundary thermal inputs to the spacecraft that are determined by the orbital environment and self-reflections/emissions may be supplied by (1) simulating incident solar and planetary irradiation (solar simulation), (2) simulating absorbed solar and planetary irradiation (IR plates, IR heater skins, IR lamp sources) or a combination of 1 and 2. Internal boundary thermal conditions may be supplied by (a) simulating mechanical and thermal mock-ups using resistance elements to duplicate internal dissipation, (b) the use of actual subsystems, or a combination of a and b.

When either simulated solar incident or absorbed flux is used for thermal balance tests, the chamber and test setup shall be calibrated to determine direction, intensity, and spectral content of energy sources and effective heat sink temperature extant in the chamber. The calibration can be obtained by direct measurement or may be derived experimentally. The resulting energy parameters selected for the test condition and the spacecraft test configuration shall be put into the spacecraft analytical model, and nodal temperatures shall be predicted for the chamber test conditions prior to the thermal balance test. If the IR absorbed flux technique is selected for the test method, separate tests shall be conducted to verify absorptivity and emissivity of the flight spacecraft thermal control surfaces.

In using the techniques described above, it is essential that thermal instrumentation be applied to the spacecraft at discrete locations corresponding to the thermal node locations in the analytic model.

Verification of thermal balance shall be considered successful if the difference between analytical predictions and measured temperatures at preselected node locations is not greater than 5°C, when the prediction is a discrete value. If temperature predictions are made as a range of values because of uncertainty in heat transfer coupling factors, then the measured value from test must fall within the range of predicted values.

The thermal balance test shall be conducted in such a manner that the spacecraft thermal analytical model is checked for at least three energy balance conditions that are most critical to the performance of the thermal control system.

(c) Operating Mode and Duration. The spacecraft shall be operated and monitored throughout the test. If necessary, the orbital mode shall be modified to achieve conditions of constant energy dissipation to facilitate model verification. For high heat dissipating subsystems which are cyclically operated during orbit (e.g. batteries, high power transmitters, etc.) transient operation shall be required to determine the maximum temperature variation for the subsystem; external boundary flux is usually held constant to facilitate analysis of the temperature effect.

The duration of the thermal balance test depends on the mission, spacecraft design, and spacecraft operating modes. For explorer and observatory class spacecraft, test durations have ranged from 6 to 25 days. The particular spacecraft specification shall include the applicable duration and shall be approved by the GSFC project manager.

(d) Documentation. The following documentation will be required at the option of the project manager.

(1) Thermal balance test plan and procedure

(2) Facility calibration plan and procedure

(3) Evaluation of test results including:

(a) A comparison of predicted performance and test performance.

(b) Adjustments required in analytical model, spacecraft design, and subsystem temperature limits.

(4) Updated thermal predictions for the mission.

(5) Temperature levels for the thermal-vacuum performance test.

2.1.13.5 Thermal-Vacuum Performance Test. This test is required as a phase of the design qualification program to evaluate and demonstrate the ability of the spacecraft to perform under the extreme and nominal modes of operation required by the mission while under simulated vacuum and temperatures more extreme than predicted. Temperature conditions for the subsystems shall be 10°C more severe than the maximum and minimum temperatures predicted for the mission. The purpose of the more severe temperature stress is to demonstrate a design safety margin and accelerate failure in marginal designs. The test shall be conducted by forcing extreme temperatures at subsystem locations by modifying the operational modes of the spacecraft, and/or by adjusting local thermal boundary conditions to provide additional heating or cooling as may be required.

The thermal-vacuum performance test shall satisfy the following minimum requirements:

(1) Corona check

(2) 108 hours operation at maximum hot temperature stress conditions

(3) 108 hours operation at minimum cold temperature stress conditions

(4) 8 transitions (while operating) between temperature extremes

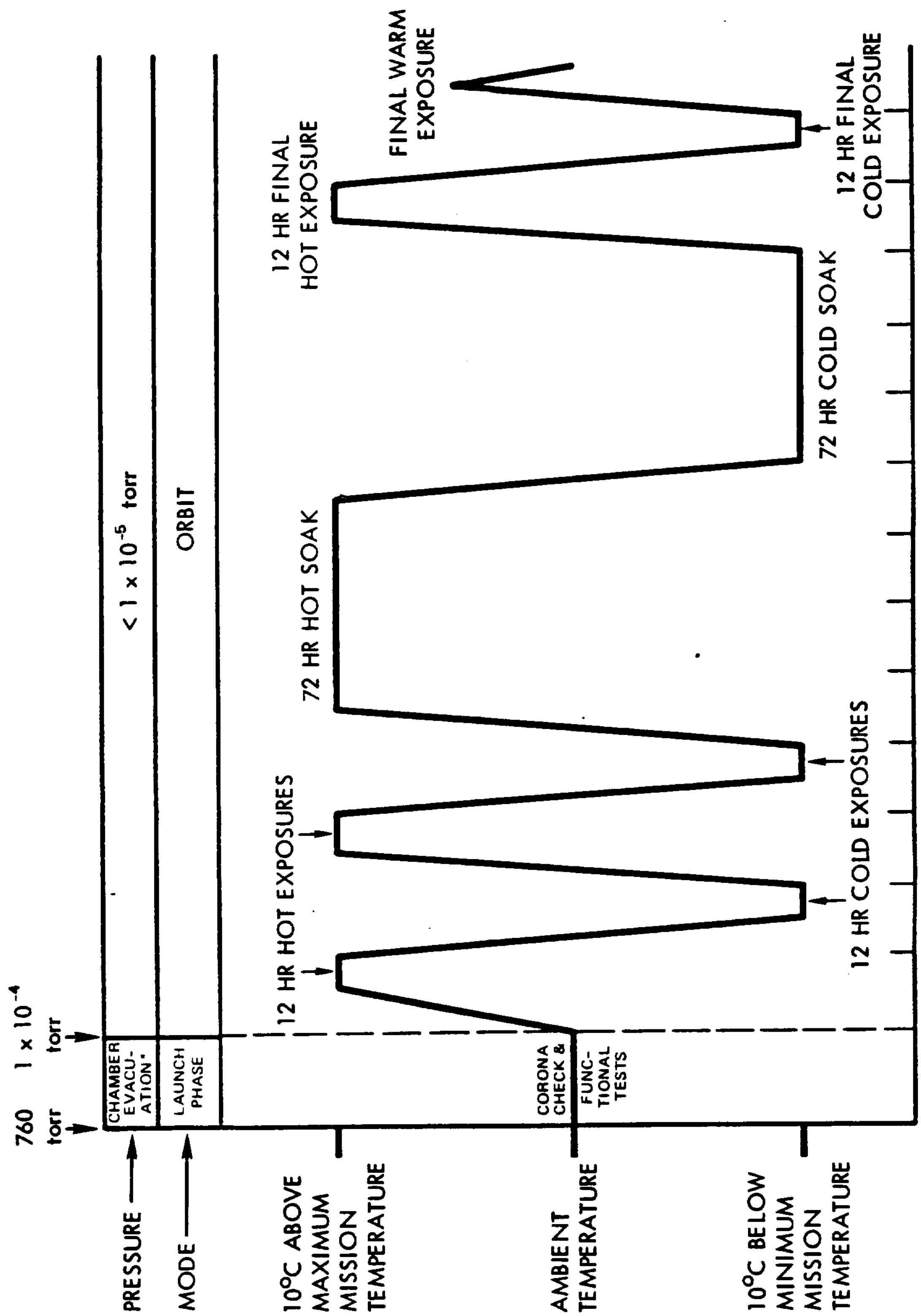
The test duration varies with spacecraft design (number of operating modes, thermal inertia, etc.) and test facility capability (thermal capacity, etc.). For explorer and observatory class spacecraft, test durations have ranged from 13 to 25 days. The particular spacecraft specification shall include the applicable duration and it shall be approved by the GSFC project manager.

A typical explorer class spacecraft test profile is shown in Figure III. That test profile subjects the spacecraft to cold and hot extremes under vacuum conditions and includes a corona check. It consists of both short tests (exposures) and longer tests (soaks). Exposures to hot and cold temperatures are made first to detect the type of failure which becomes evident early, before undertaking the more time-consuming soaks. The purpose of the soaks is to uncover weaknesses which become evident only during prolonged periods of extreme temperature and vacuum conditioning.

A final hot and cold exposure after the cold soak has the purpose of uncovering any degradation which may have occurred in the hot and cold soak but would become evident only in a subsequent hot-cold cycle. The cycling between temperature extremes is to induce temperature gradient shifts in the spacecraft, thereby permitting observation of operation at other than stabilized conditions. The purpose of the corona check is to verify that no permanently damaging electrical discharge occurs during transition to vacuum conditions. Spacecraft cold start-up capability is demonstrated at cold conditions to verify that the spacecraft will turn on should minimum temperatures be attained in orbit. The spacecraft system shall be either in a commanded-off or undervoltage-recycle mode.

During transition periods and periods of temperature stabilization, the spacecraft shall be operated in its normal orbital modes. Stabilization is determined by monitoring the control sensors selected prior to the test. The test chamber wall temperature shall be adjusted as necessary to maintain the control sensor at the desired level.

The spacecraft shall be in as near an orbital configuration as possible for the thermal-vacuum test. The test



*Chamber evacuation takes about 2 hours so time scale is disproportionate here.

**The plot shows 12 hours for each temperature transition; actually, the transition periods vary widely with spacecraft, mission, and test equipment.

Figure III–Typical Thermal-Vacuum Test Cycle for Prototype Spacecraft.

setup for the corona check shall be reviewed to insure that the test is valid for both the launch and orbital configurations. The final test configuration shall be approved by the GSFC project manager.

(a) Corona. Upon completion of the test setup, per 2.1.8.4, the spacecraft shall be operated in its launch phase mode while the chamber is being evacuated to a pressure of 1×10^{-4} torr at ambient temperature. The time to reach 1×10^{-4} torr shall follow the launch pressure time profile where practicable. When a pressure of 1×10^{-4} torr is achieved, the spacecraft shall be switched to its normal orbital operating mode and the test shall be continued into its next phase. The purpose of continuing the corona check into the next phase is to permit the pressure to decay in those components which may have a pressure-time lag. The spacecraft shall be monitored for corona and other high-voltage phenomena during this test.

(b) Hot Exposure. Following the corona check, the chamber shall be evacuated until a pressure of less than 1×10^{-5} torr is achieved in the chamber working volume. The chamber temperature shall be increased to cause the spacecraft to stabilize at the hot exposure test temperature.

This temperature shall be 10°C above the maximum mission temperatures established by a survey of the predicted temperatures for each of the components/subsystems/experiments except solar paddles and boom mounted sensors and skins. If this maximum component/subsystem/experiment temperature would be damaging to the performance of other subsystems, the temperature of the warmest component shall be controlled individually during the test and the next lower component temperature found in the survey shall be selected as the test level control point. If necessary, components with successively lower temperatures shall be selected as the basis for the hot stabilization control point for conditioning the bulk of the spacecraft while all components requiring warmer temperatures shall be individually controlled.

This procedure usually results in separate thermal conditioning for appendages and external skin-mounted components. Special conditioning for components within the spacecraft shall be avoided if possible.

The duration of this test is twelve hours after the control sensor reaches the test level. If more time is required to complete the cycle of orbital operations, the twelve-hour period shall be extended accordingly.

(c) Cold Exposure. After completion of the hot exposure, the wall temperature of the chamber shall be decreased to cause the spacecraft to stabilize at the cold exposure test temperature. Cold turn-on capability shall be demonstrated at the start of each cold exposure.

The temperature shall be 10°C below the minimum temperature established by a survey of the predicted temperatures which each of the components/subsystems/experiments would experience during the mission. If this minimum component/subsystem/experiment temperature would be damaging to the performance of other subsystems, the temperature of the coldest component shall be controlled individually during the test and the next higher component temperature found in the survey shall be selected as the basis for the test control point temperature. If necessary, components with successively warmer temperatures shall be selected as the basis for the cold stabilization wall temperature for conditioning the bulk of the spacecraft while all components requiring colder temperatures shall be individually controlled. Special conditioning of components within the spacecraft shall be avoided, if possible.

The duration of the cold exposure test shall be twelve hours after the control sensor reaches the test level. If more time is required to complete the cycle of orbital operations, the twelve-hour period shall be extended accordingly.

(d) Hot Exposure. After completion of the cold exposure above, the test procedure for the hot exposure (paragraph b) shall be repeated.

(e) Cold Exposure. After completion of the hot exposure above, the test procedure for the cold exposure (paragraph c) shall be repeated.

(f) Hot Soak. After completion of the cold exposure above, the test procedure for the hot exposure (paragraph b) shall be repeated except that the test duration shall be 72 hours instead of 12.

(g) Cold Soak. After completion of the hot soak above, the test procedure for the cold exposure (paragraph c) shall be repeated except that the test duration shall be 72 hours instead of 12.

(h) Final Hot Exposure. After completion of the cold soak above, the test procedure for the hot exposure (paragraph b) shall be repeated.

(i) Final Cold Exposure. After completion of the hot exposure above, the test procedure for the cold exposure (paragraph c) shall be repeated.

(j) Final Warm Exposure. After completion of the final cold exposure, the spacecraft temperature shall be elevated and maintained above ambient temperature while the chamber wall is being returned to ambient. This procedure is recommended to reduce the risk of molecular contamination of spacecraft surfaces.

2.1.14 Antenna Pattern Determination

2.1.14.1 Requirements. The spacecraft or a mockup shall be subjected to a comprehensive check on the antenna range to accurately determine the pattern of each spacecraft antenna. Results of the check shall be documented and must be within design limits.

2.1.14.2 Mock-up. A full-sized mock-up of the spacecraft may be used with the actual antennas for this test if the conducting and nonconducting surfaces on the mock-up occupy the same relative positions as on the spacecraft. Also the dielectric constant of the nonconducting surfaces on the mock-up should approximate those of the spacecraft.

It is permissable to use scale model spacecraft for antenna pattern determination if sufficient care is employed.

2.1.15 Electromagnetic Interference (EMI)

2.1.15.1 General Requirements. The spacecraft electrical and electronic equipment should operate satisfactorily not only as independent systems but also in conjunction with the launch vehicle and ground support equipment and in the proximity of launch range equipment. In short, the spacecraft should not be adversely affected by electromagnetic interference reaching it from any external sources. Conversely, the spacecraft itself should not be a source of interference which might adversely affect its own operation, vehicle operation, other spacecraft, or ground monitoring and control equipment.

The spacecraft shall be subjected to the required electromagnetic tests while in both launch and orbital configurations and in all normal operational modes.

Electro-explosive devices (EED) with bridge-wires, but otherwise inert, shall be installed in the spacecraft during all tests.

2.1.15.2 Modification for Particular Spacecraft. After consideration of the above general requirements and the characteristics of the particular spacecraft, the following provisions may be modified by the project manager and the modified version shall be included in the particular spacecraft specification.

2.1.15.3 Radiated Electromagnetic Interference Tests. Broadband and CW interference tests shall be conducted in accordance with MSFC-SPEC 279. The ambient electromagnetic level of the test area shall not be high enough to affect EMI measurements made on the spacecraft. The bands or frequencies for the required tests shall be scanned slowly enough to ascertain discrete frequencies or bands of maximum interference. At least three measurements per octave shall be recorded. An X-Y plotter (8 x 11 report size) for peak detection shall be used to record the signature display of the bands of interference.

2.1.15.4 Broadband Radiated Interference Tests. Measurements shall be performed over the frequency range of 14 kHz to 400 MHz with the measuring antenna(s) positioned 1 foot from the nearest periphery of the spacecraft and located to produce maximum interference pickup. Readings shall be made in db above one microvolt per megahertz bandwidth. Radiated interference data shall be derived from test instrument readings plus the appropriate antenna and cable correction factors.

2.1.15.5 CW Radiated Interference. Measurements shall be performed over the frequency range of 15 kHz to 10 GHz* with the measuring antenna(s) positioned 1 foot from the nearest periphery of the spacecraft and located to produce maximum interference pickup. Each octave is scanned prior to the measurement and the highest amplitude CW signals is measured and recorded. Readings are made in db above one microvolt. Radiated interference data are derived from test instrument readings plus the appropriate antenna and cable correction factors.

2.1.15.6 Conducted Electromagnetic Interference Tests. Broadband and narrowband conducted interference tests shall be performed to measure the existence and levels of interference signals on power and signal lines. Tests shall be conducted in accordance with MSFC-SPEC-279

*The 1 to 10 GHz range need not be measured unless there is reason to suspect narrowband interference signals.

in an ambient electromagnetic background level which will not interfere with desired measurements. Frequency band scanning and interference signal recording techniques as in 2.1.15.5 shall be used. The tests are as follows:

(a) Power-Line Conducted Interference. Line Impedance Stabilization Networks (LISN) shall be used in this test to determine the levels of broadband and narrowband conducted interference in the frequency range of 0.15 to 25 MHz. Broadband readings shall be made in db above one microvolt per megahertz bandwidth. Narrowband readings shall be made in db above one microvolt.

(b) Power-Line and Signal-Line Conducted Interference With Current Probe. This test is to determine the levels of broadband and narrowband conducted interference in the frequency range of 30 Hz to 25 MHz. Narrowband readings shall be in db above one microampere. Broadband readings shall be made in db above one microampere per megahertz bandwidth.

2.1.15.7 Electromagnetic Susceptibility Tests. MSFC-SPEC-279 shall be used for electromagnetic susceptibility tests. Tests shall be performed by illuminating the spacecraft with electromagnetic signals over the frequency range of 0.15 MHz to 10 GHz. The threshold where degradation or malfunction occurs shall be measured in volts/meter field strength at the nearest periphery of the spacecraft for all normal operating modes. Any malfunction or degradation of spacecraft performance shall warrant further investigation of the particular frequencies involved. In such cases, the transmitting signal generator output shall be lowered to the point where no degradation occurs in order to quantitatively evaluate equipment susceptibility. Interference shall be at least 6 db below the above established thresholds.

2.1.15.8 Test Report. A test report shall be prepared containing complete records of the EMI tests. These shall include data sheets, performance curves, photographs or drawings of layout and location of equipment and test specimen, schematics of wiring, sample calculations of interference signal levels, and a description of the test equipment and procedures in sufficient detail so that the test can be duplicated and evaluated by persons not witnessing the test. Calibration dates of test equipment shall be provided. The report shall also contain signal and power-line impedance characteristics, line length data, and other information where necessary to aid in the design of filters and/or shielding at the appropriate interfaces of subassemblies when these are required to eliminate EMI problems.

2.1.16 Operational Spin and Mechanical Functioning

2.1.16.1 General Requirements. Operational spin and mechanical functioning tests of the spacecraft in its launch and orbital configurations shall be conducted to confirm spacecraft performance and to assure that no degradation has occurred during environmental testing. During the tests, the electrical and mechanical systems of the spacecraft shall be in the various operational modes appropriate for launch and orbital configurations. For electrical systems and pyrotechnic devices the provisions of 2.1.3 and 2.1.4 for performance testing shall apply. Mechanical spacecraft systems and those portions of the spacecraft launch configuration which, although not a part of the spacecraft, affect launch phase operation shall be subjected to functional tests to demonstrate the adequacy of specific mechanical system design.

In addition to loads due to mechanical functioning, the spacecraft shall be exposed to pertinent environmental effects prior to and during the operational test. These are referred to as environmental preconditioning and conditioning, respectively (2.1.16.4). The particular spacecraft specification shall stipulate the tests to be conducted, environmental conditioning, and the range of required operation.

It is mandatory that all tests be witnessed by qualified observers.

To uncover weaknesses, limiting phenomena, failure modes and mechanisms, and to develop and institute quality control procedures, it is desirable to perform preliminary mechanical systems tests and exploratory design development tests early in the design qualification program using a structural model of the spacecraft.

2.1.16.2 Typical Mechanical Subsystems. Systems to be tested shall include, but are not limited to: spin up; shroud ejection or repositioning; clamp actuation; spacecraft and/or stage separation; despin; paddle and boom erection, extension, and repositioning; appendage cradle operation; and release mechanisms. Whenever possible, flight type hardware shall be subjected to the tests. If flight type hardware cannot be used, test hardware shall duplicate flight hardware in all essential respects. This especially includes the dynamic effects of mass and its distribution, damping, compliance, fit, and friction.

2.1.16.3 Subsystem Data Requirements. The following information is required to define the specific test parameters.

(a) Mission requirements for subsystem operation. A description of the subsystem function, how the system is intended to operate, and when it occurs in the overall spacecraft mission.

Required range of acceptable operation, such as operation and/or survival over a range of spin speeds, temperature, etc. Also associated with this are criteria of acceptability and measurements that are required to confirm success.

(b) Flight conditions. Anticipated variation of all pertinent flight conditions or system parameters which may affect system performance. Included in this as applicable, are: weight, inertias, spin rates, final position, separation and tip off rates, coning, temperature, and long-time set (storage).

2.1.16.4 Environmental Simulation.

(a) Conditioning. If necessary, on the basis of analysis or prior component or subsystem tests, the spacecraft shall be preconditioned before test or conditioned during test to pertinent (usually flight acceptance) environmental test levels. This can include, especially, factors of vibration, high and low temperature cycling, vacuum pressure-time profiles, and test and transportation handling.

As a caution, when retesting or testing at more than one level of severity is required, environmental reconditioning may be necessary for each test. If the failure mechanism is well understood and has been fully evaluated, consideration may be given to testing without reconditioning in the sequence which will furnish the desired evaluation of limit level performance.

(b) Reduction of air and gravity effects. During the operational test, the loads and restraints of air and gravity which would be different from flight shall be reduced so as not significantly to affect the functional performance. As a guide, a vacuum of one torr is satisfactory; for g-negation, see section 1.10h.

Overspin as a technique for overcoming gravity potential during spin deployment testing may be utilized but must be used with caution since it introduces many unrealistic structural loads. However, its simplicity as an exploratory technique makes it useful for preliminary test of single-level, single-hinged booms.

2.1.16.5 Test Procedure. With all appropriate systems functioning, the spacecraft shall assume in proper sequence the configurations of launch and flight, starting with the initial launch configuration. Spin up, separation, despin, appendage erection, deployment, retro-motor separation and/or other mechanical operations shall be accomplished as appropriate for the particular spacecraft. Most mechanical subsystems require a series of three operational tests as set forth in 2.1.16.6. One of these tests may have been accomplished during the sequence of operational tests prescribed above. Spacecraft performance should be checked during all tests.

2.1.16.6 Test Levels. For mechanical operations such as multiple hinge or multiple height paddle deployment, three tests usually are required. The levels for these tests are developed below.

One test, ordinarily the first except as explained in 2.1.16.4, should be conducted at the level most probable during normal flight (the nominal level). The reasons for this test are:

1. To establish that spacecraft functioning is proper for nominal operating conditions, and

2. To obtain measurements to serve as a basis for comparison with the nominal test on the flight spacecraft.

Other tests shall be conducted to prove positive margins of strength and function. An overtest (high level) and an undertest (low level) should be conducted. The levels of these tests shall demonstrate margins:

1. Beyond the specified operational limits, or

2. Beyond nominal operational levels if a range is not specified.

The margins should not be selected arbitrarily, but should take into account all the uncertainties of operation, strength, and test.

These uncertainties include, but are not restricted to: (1) test instrumentation and g-negation error, (2) friction variations, including effect of alignment, fit, wear, and lubrication, (3) cable harness restraint, including effect of vibration, long-time set, coupled with temperature during set and during test (unless flight conditioning is duplicated), (4) hardware, including material and manufacturing tolerances, and (5) reduction of power or drive forces due to relaxation of strained members, leakage of gases, or diminution of electrical power.

(a) General method for establishing "high" and "low" test levels. The test levels shall be the limit values required by 2.1.16.3 or the nominal operating level, multiplied by a cumulative uncertainty factor (CUF). Figure IV illustrates this concept.

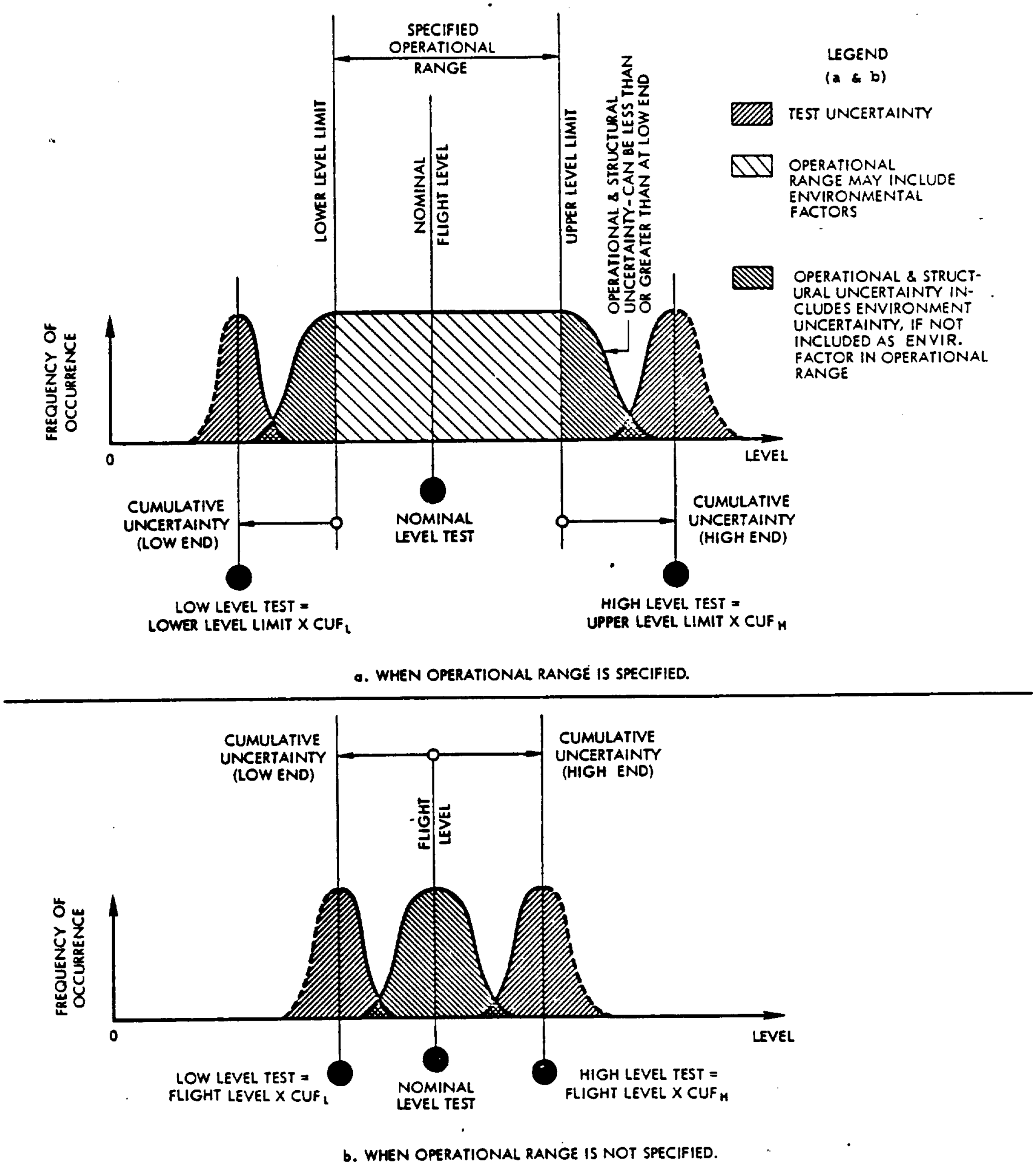


Figure IV–Determination of Operational Test Levels.

(1) High level test. For the high level test, the cumulative uncertainty factor for the high end (CUF_H) is the product of separate quantities, each of which consists of unity plus the particular uncertainty factor expressed as a decimal. Uncertainties applicable to the high end are to be used. Following is an example for determining CUF_H, using appropriately assumed uncertainties:

4%: decrease in friction due to lubrication

±5%: variation in material strength

+10%-3%: manufacturing size tolerance

±12%: test error

Then the CUF_H would be

1.04 x 1.10* x 1.13** x 1.12 = 1.45

(2) Low level test. For the low level test, the cumulative uncertainty factor for the low end (CUF_L) is the product of separate quantities, each of which consists of unity minus the particular uncertainty factor expressed as a decimal. Uncertainties applicable to the low end are to be used. The following is an example for determining CUF_L, using appropriately assumed uncertainties.

15%: increase in friction due to lack of lubrication

6%: additional increase in friction due to wear

±10%-3%: manufacturing size tolerance

±12%: test error

*Because a material 5% stronger may be tested and one 5% weaker may be flown.
**Because a part 10% larger may be tested and a part 3% smaller may be flown.

Then the CUF_L would be

$0.85 \times 0.94 \times 1.00^* \times 0.88 = 0.70$

(3) Test levels for proto-flight spacecraft. The same principle for determining the prototype CUFs applies to proto-flight** spacecraft except that in computing the CUFs it should be recognized that some of the uncertainties might not exist, such as: variation of friction due to lubrication and manufacturing size and finish; manufacturing differences of material strength and component size, etc. Accordingly, the CUFs would result in less deviation from nominal or from limit levels.

It is particularly important to develop measurement, evaluation, and quality control techniques to assure that degradation due to the testing will not preclude the actual flight mechanical function.

(b) Simplified method for establishing levels, and when operational range is not specified. If a range of acceptable operation has not been specified, one nominal level test shall be run to serve as a basis for comparison. A minimum of two qualification level tests shall be conducted to demonstrate that no yield or degradation has occurred. The tests shall also serve to demonstrate a positive capacity to perform the desired function. Relative to the so-called nominal condition, one test shall be at a level which imposes increased energy load, thereby demonstrating strength; the other shall apply less energy or load, or more restraint, to demonstrate excess capacity.

If the overall error of the test measurement and simulation is less than ±15%, and if the quality control of the hardware limits the hardware variables to ±15%, and if there are no unknown or special

*Not 1.00 + 0.10 or 1.00 - 0.03, as change in size is not directly related to function.

**See 4.11 for definition of proto-flight spacecraft and discussion of overall environmental test program requirements.

degradation factors, the qualification test levels shall be at 1.5 and 0.5 times nominal.

For proto-flight models, the tests shall be at 1.4 times, at 0.6 times, and at nominal levels. The note of caution on the development of quality control techniques as stated three paragraphs above also applies.

2.1.16.7 Level Control. If knowledge of the failure mechanism is well understood, any factor which realistically could vary due to environmental exposure or from spacecraft operational functioning, and which can be readily and predictable controlled during test, may be used to impose the desired test levels. Parameters which have been successfully varied are: spin speed; for example at .75 and 1.25 times nominal for 44% undertest and 56% overtest of load, respectively; driving force or load at .50 and 1.50 times nominal; booster springs, and overcompensation or under-compensation for gravity.

If load limiters are part of the design and they prevent a proper strength overtest, the stress, load, or moment developed at and by nominal operation shall be measured. The load at the desired level shall then be applied quasi-statically* with the limiter locked out of operation. An alternative to the quasi-static test is to verify strength of critically loaded members by stress analysis techniques. The indicated margin of excess strength must be consistent with the criticality of the member as well as the accuracy of the analysis.

2.1.16.8 Criteria of Acceptance and Measurement of Performance. That which constitutes acceptable performance in individual cases should be specified by the spacecraft designer and project manager in the particular spacecraft specification. Acceptable performance may vary considerably with the spacecraft. Repeatability of position with no yield may be required; or a specified amount

*Quasi-statically means that the rate of load buildup is less than that which would cause appreciable dynamic effects, and that the duration of load application shall be minimized, consistent with reaching test load.

of yield may be allowed if it does not interfere with or degrade the mechanical system, spacecraft rpm, separation velocities, or exceed specific tip-off conditions.

As a minimum, the detailed test plan and procedure, approved by the project manager, shall list the acceptance criteria and the measurement which will be taken to verify performance.

Some useful measurements are:

(a) Initial and final position before and after dynamic functioning to show repeatability and yield. Repeatability of final position is not only an indicator of strength; the position measurements after nominal level tests are essential inputs for analytical balancing operations; see 2.1.5.3 and 2.2.4.2.

(b) Position versus time, velocity, or acceleration to demonstrate uniformity of motion, and especially that no unanticipated changes of acceleration occur, such as jerk. Uniformity of motion is a good indication of proper operation.

(c) Acceleration of experiment packages so that the data can be used to develop levels for qualification tests of those components.

(d) Stress or moment in structural members; for failure analysis, study of loading dynamics, determination of strength margins, and for use in quasi-static testing.

(e) Stop motion photography, high speed photography, and television with instant replay are useful diagnostic aids, especially if there is little background information or experience, or if full dynamic or structural analysis is impracticable or unavailable.

To minimize change of perspective, camera view angles should be at right angle to the plane of expected motion. It is useful if the spacecraft has been designed in such manner that the deployed appendages provide an exaggerated optical indication of their position.

2.1.16.9 Separate Testing of New Mechanical Subsystems. It is important that new mechanical systems be tested as systems early in the spacecraft test program utilizing the most complete environmental simulation available. At that time, degrading interactions which may not have been anticipated can be disclosed and corrected.

2.1.17 Final Magnetic Field Measurement. After the spacecraft has undergone the specified environmental test program, it shall be subjected to a magnetic field measurement as required by the particular spacecraft specification to determine the effect of such tests on the permanent, induced, and stray magnetic moments of the spacecraft. The spacecraft shall be positioned inside the magnetic test coils via a fixture of nonmagnetic material. If necessary, the spacecraft shall be subjected to a deperm treatment to reduce such moments to within the limits previously measured in the initial magnetic field measurement.

2.2 SPACECRAFT FLIGHT ACCEPTANCE

Flight acceptance tests subject flight spacecraft to environmental levels equal to those expected in ground handling, launch, and orbit. The purpose of these tests is to locate latent material and workmanship defects in a proven design.

Successful completion of this test sequence results in acceptance of the flight spacecraft for launch.

2.2.1 Leak Detection. The provisions of 2.1.2 for leak check of prototype spacecraft shall apply to flight spacecraft except for pressure of sealed specimens in the "Sniff" test. Tables VIII and IX contain all parameters for leak checking of flight spacecraft. Also the requirement in 2.1.2 for leak checks before and after the temperature and humidity test phase does not apply to the flight acceptance test program.

TABLE VIII

LEAK DETECTION VACUUM TEST SCHEDULE
SPACECRAFT FLIGHT ACCEPTANCE

Applicable To	Pressure of Sealed Specimen	Proportion of Tracer Gas (Pressure)	Chamber Pressure	Maximum Leak Rate*
Sealed spacecraft or space-craft with sealed units	100 to 760 torr	100%	1×10^{-4} torr	1×10^{-6} atm std cc/sec.
	760 to 1520 torr (Absolute)	10 to 100%		

*Or as otherwise established by spacecraft design. Leak rate of pressurized gas attitude control systems during non-operating phases of the operational cycle shall not exceed the maximum leak rate established by design limits.

TABLE IX

LEAK DETECTION NON-CHAMBER "SNIFF" TEST
SPACECRAFT FLIGHT ACCEPTANCE

Applicable as Required To	Pressure of Sealed Specimen	Proportion of Tracer Gas (Pressure)	Function
Sealed spacecraft or spacecraft with sealed units	760 torr (Absolute)	100%*	Determines leak locations, not leak rate

*Desirable

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2.2.2 Electrical Performance Test

2.2.2.1 Purpose. The purpose of this test is to verify electrical performance of all systems during the spacecraft flight acceptance test program. Satisfactory electrical performance before, during, and after the specified environments shall be required prior to acceptance of the flight spacecraft for launch.

2.2.2.2 Requirements. The provisions of 2.1.3 for electrical performance testing of prototype spacecraft shall apply to flight spacecraft. Such testing shall be conducted at the beginning of the flight acceptance test program as well as before, during, and after each exposure in the spin, vibration, thermal radiation/thermal-vacuum and solar simulation tests, as specified in the provisions for those tests.

2.2.2.3 Procedure. Although the flight spacecraft is constructed to duplicate the physical aspects and capabilities of the prototype spacecraft, experience has shown that electrical performance characteristics are not duplicated. Therefore, the reference data established for electrical testing of the prototype spacecraft cannot be used for the flight model, and independent reference data must be established.

2.2.3 Pyrotechnic Performance. Same as provisions for prototype spacecraft in 2.1.4.

2.2.4 Balance - Initial

2.2.4.1 General Requirements. Balance operations shall satisfy requirements for launch and orbital configurations. The spacecraft shall be balanced while in a non-operative state. To correct unbalance, weights shall be attached, removed and/or relocated as approved by the designated representative of the project manager. Necessary spacecraft modifications, including optimizing the location of components, shall be with the approval of a designated representative of the GSFC project manager. The amount of residual unbalance for both launch and orbital configurations shall be measured and recorded for comparison with the specification balance. The spin rate used in balancing any configuration of the spacecraft shall not normally exceed that expected in flight. Balance operations shall include interface fit and alignment checks as necessary to ensure alignment of geometric axes compatible with balance requirements.

2.2.4.2 Analytical Balancing. Balancing operations shall include measurement and tabulation of physical parameters (weight and mass center location referenced to spacecraft coordinates) of system elements which may not be assembled for spin balancing. This data shall be processed to determine unbalance contributed by these elements to launch and orbital configurations. Measurement techniques shall include compensation and/or correction for the effects of yield (see 2.1.16.8), assembly tolerances, spin and gravity on the accuracy, and repeatability of the measurements. Unbalance due to imperfect control of these factors should not exceed 50% of the applicable final flight acceptance balance specification per Table 10 of the applicable launch vehicle appendix.

The schedule for required analytical balancing operations is discretionary, subject to efficient attainment of the final flight acceptance balance specification.

2.2.4.3 Launch Configuration. The vehicle design restraints manual applicable at time of test forms the basis for the launch configuration balance requirements. Requirements for initial balance of the flight spacecraft allow one and one-half the unbalance permitted in the final balance of the flight spacecraft and are stated in Table 6 of the applicable launch vehicle appendix.

2.2.4.4 Orbital Configuration. Orbital balance requirements, based on the particular spacecraft mission, shall be furnished by the project manager. Balance requirements and procedures shall appear in the particular spacecraft specification.

2.2.5 Weight, Center of Gravity, Moments of Inertia. Same as provisions for prototype spacecraft as stated in 2.1.7. Measurements may be made after final balance, or adjusted to reflect the best data available at that time.

2.2.6 Spin

2.2.6.1 When Conducted. This test is not normally conducted but the project manager may require it for new or unproved designs or when the design qualification spin test has shown an unusually high sensitivity to failure.

2.2.6.2 Requirements. The provisions of 2.1.6 for spin-testing the prototype spacecraft shall apply except that the spin rates shall be at predicted nominal speed as shown in Table X.

TABLE X

SPACECRAFT FLIGHT ACCEPTANCE SPIN

Electrical Operation	Spin Rate	Duration (min)
All applicable systems (2.1.6.2)	Nominal launch or orbital rate (whichever is greater)	10*

*Longer if necessary to verify spacecraft operations

2.2.7 Vibration

2.2.7.1 General Requirements. The provisions of 2.1.9 for vibration testing of prototype spacecraft apply to flight spacecraft except for test parameters. The following paragraphs and Tables 7 and 8 of the applicable launch vehicle appendix present the parameters for testing flight spacecraft. If only a single, or proto-flight, spacecraft exists, it shall be treated according to provisions of 2.1.9.2.

2.2.7.2 Sinusoidal Vibration. The sinusoidal portion of the test shall be performed by sweeping the applied frequency once through each range specified in Table 7 of the applicable launch vehicle appendix.

2.2.7.3 Random Vibration. Gaussian random vibration shall be applied for each axis specified in Table 8 of the applicable launch vehicle appendix. With the spacecraft installed, the control accelerometer(s) response shall be equalized such that the power spectral density (PSD) values, as determined by the analysis specified in 2.1.9.12, are within ±3 db throughout the frequency band and the overall rms level is within ±10% of that specified. The filter roll-off characteristics above 2000 Hz shall be at the rate of 40 db/octave or greater.

2.2.8 Acoustic Noise. The provisions of 2.1.10 for acoustic noise tests of prototype spacecraft shall apply also to flight spacecraft except for the test parameters listed in Table 9 of the appropriate launch vehicle appendix.

2.2.9 Shock. The provisions of 2.1.11 for shock testing of prototype spacecraft, shall apply also to spacecraft flight acceptance testing except that each type of shock need be applied only once. The programmed shock, if required, is defined by the shock spectrum figure in the applicable launch vehicle appendix.

2.2.10 Space Environment Operation Check

2.2.10.1 Purpose. The purpose of this test is to obtain assurance that the spacecraft is capable of operating successfully under conditions representative of those which it will encounter in space. The provisions of 2.2.2 for electrical performance testing shall apply.

Thermal stress conditions for this test shall be based upon analytic and experimental refinements resulting from the thermal balance design qualification test program such that the flight spacecraft is subjected to predicted thermal extremes. If the thermal balance test has been deferred until the flight spacecraft acceptance test, the requirements of section 2.1.13.4 and 2.2.10.2c shall apply.

Representative conditions can be produced in the laboratory by either of two different methods: Method 1, by simulating the predicted incident or absorbed heat flux in vacuum (space thermal radiation); or Method 2, by exposing the spacecraft to the predicted temperature extremes in vacuum (thermal-vacuum test). When Method 2 is used, separate tests shall be required to verify successful duplication of the proven thermal balance design (2.1.13.4) and the absorptivity emissivity ratio of the thermal control surfaces of the coated spacecraft. The particular spacecraft specification shall prescribe the method to be used. In either case, a corona check shall be conducted immediately prior to the test to verify that no significant electrical discharge occurs during transition to vacuum conditions.

2.2.10.2 Space Thermal Radiation. If the space thermal radiation test (Method 1) has been selected for conducting the flight acceptance test, it shall immediately follow a corona check per 2.1.13.5a. The chamber shall continue to be evacuated until a pressure of less than 1×10^{-5} torr is achieved in the chamber working volume. In this vacuum the integrated spacecraft shall be subjected to thermal radiation fluxes as stipulated below.

(a) Extreme Mean Spacecraft Temperatures. The radiation intensities shall simulate the coldest and hottest energy balance extremes predicted for the mean spacecraft temperature during the mission (see 2.2.10.1).

(b) Spacecraft Thermal Gradients. If there are unusual spacecraft thermal gradients during the mission which cause certain components/subsystems/experiments to run 10°C hotter or colder than the temperatures they will experience during the mean spacecraft temperature extremes, conditions shall be established in the chamber which will subject these particular subsystems to their expected temperature extremes. This result may be accomplished by local thermal conditioning of the particular component or by subjecting the entire spacecraft to thermal inputs which produce the desired thermal gradients in the spacecraft.

(c) The duration of the acceptance test required by 2.2.10.2 shall be sufficient to accomplish the following minimum stress conditions for the subsystems.

(1) Corona check

(2) 108 hours operation at maximum hot temperature stress conditions

(3) 108 hours operation at minimum cold temperature stress conditions

(4) 8 transitions (while operating) between temperature extremes

2.2.10.3 Thermal-Vacuum Test. If the thermal-vacuum test (Method 2 for conducting the flight acceptance test) is selected, all of the provisions of 2.1.13.5 shall apply except that the extreme cold and extreme hot temperatures shall be based on the expected mission extremes in 2.2.10.1 instead of 10°C below and above the mission extremes. The duration of the test shall be as required to meet the minimum exposure conditions per 2.2.10.2c.

2.2.11 Balance - Final

2.2.11.1 General Requirements. Balance operations shall satisfy requirements for launch and orbital configurations. The spacecraft shall be balanced while in a nonoperative state. To correct unbalance, weights shall be attached, removed and/or relocated as approved by the designated representative of the project manager. Necessary spacecraft modifications, including optimizing the location of components, shall be with the approval of the spacecraft contractor and a designated representative of the GSFC project manager. The amount of residual unbalance for both launch and orbital configurations shall be measured and recorded for comparison with the specification balance. The spin rate used in balancing any configuration of the spacecraft shall not normally exceed that expected in flight. Balance operations shall include interface fit and alignment checks as necessary to insure alignment of geometric axes compatible with balance requirements.

2.2.11.2 Analytical Balancing. Balancing operations shall include measurement and tabulation of physical parameters (weight and mass center location referenced to spacecraft coordinates) of system elements which may not be assembled for spin balancing. This data shall be processed to determine unbalance contributed by these elements to launch and orbital configurations. Measurement techniques shall include compensation for the effects of yield, assembly tolerances, spin and gravity on the accuracy and repeatability of the measurements. Unbalance due to imperfect control of these factors should not exceed 50% of the applicable final flight acceptance balance specification per Table 10 of the applicable launch vehicle appendix.

The schedule for required analytical balancing operation is discretionary, subject to efficient attainment of the final flight acceptance balance specification.

2.2.11.3 Launch Requirements. The launch vehicle spacecraft design restraints manual applicable at time of test prescribes the final balance requirements for the launch configuration. Except as modified below, the values are those specified in Table 10 of the applicable launch vehicle appendix.

For spacecraft launched in spin stabilized stages, the final criteria of spacecraft balance acceptability is the static and dynamic unbalance contributed to the final stage/spacecraft assembly. A significantly larger spacecraft dynamic unbalance than specified in Table 10 is acceptable in most cases. This relaxation is permitted if the static and dynamic unbalance contributed by the spacecraft to the third stage/spacecraft assembly can be shown to be less than the static and dynamic unbalance which would result if the static and dynamic unbalances allowed by Table 10 existed and were in phase and additive.

2.2.11.4 Orbital Requirements. The particular test specification prescribes the balance requirements for the orbiting spacecraft. When feasible, however, these orbital values shall be halved to provide a margin of safety for disassembly, substitution, etc., at the launch site.

2.2.11.5 Monitoring of Spacecraft Balance after Final Balance. The effect on balance of any disassembly, substitution or other operation which may occur between final balance and launch requires appropriate evaluation. The project manager shall assign responsibility for maintaining a complete record and evaluation of actions affecting spacecraft balance.

2.2.12 Antenna Pattern Determination. Same as for prototype spacecraft, 2.1.14.

2.2.13 Electromagnetic Interference. Same as for prototype spacecraft, 2.1.15.

2.2.14 Operational Spin and Mechanical Functioning. The provisions of 2.1.16 for testing the prototype and structural model spacecraft shall apply to flight spacecraft except that the test level shall be at nominal environmental and operational conditions. The results shall be correlated with the "nominal" test which was performed under the qualification test program.

Because of risk to certain hardware, it may be necessary to substitute equivalent structural test hardware. For the same reason, or because of the "one-shot" nature of a device, it may be necessary to substitute quality control checks or statistical methods to insure that the flight mechanical system is acceptable. These modifications must be mutually acceptable to the project manager and the Director of Systems Reliability.

2.2.15 Final Magnetic Field Measurement. Same as final magnetic field measurement for prototype spacecraft, 2.1.17. For missions where correct spacecraft performance is sensitive to magnetic moments, care should be taken to maintain magnetic cleanliness and the validity of final measurements in the period between final deperm and launch.

SECTION 3

TESTING OF COMPONENTS
(Includes Subsystems and Experiments)

3.1 COMPONENT DESIGN QUALIFICATION

3.2 COMPONENT FLIGHT ACCEPTANCE

3.1 COMPONENT DESIGN QUALIFICATION

Design qualification tests for components are intended to simulate conditions which are somewhat more severe than ground, launch, and orbital conditions in order to locate design deficiencies; however, the conditions are not intended to be severe enough to exceed design safety margins or to induce unrealistic modes of failure. Should such modes occur, pertinent requirements may be waived.

The separate testing of components as set forth in this section may be waived by the project manager.

The provisions of this section apply to subsystems and experiments as well as components.

3.1.1 Physical Measurements and Center of Gravity. The weight and center of gravity of components shall be determined in accordance with the requirements and tolerances specified by Table XI.

TABLE XI

WEIGHT DETERMINATION
COMPONENT DESIGN QUALIFICATION

Applicable To	Parameter	Tolerance
Components, Experiments, Subsystems	Weight	±0.1% or ±.01 lb, whichever is greater
	Center of Gravity	±.06 in.

3.1.2 Initial Magnetic Field Measurement. The provisions of 2.1.1 for magnetic measurement of spacecraft also shall apply to components.

3.1.3 Leak Detection. The provisions of 2.1.2 for prototype spacecraft leak detection shall apply to components which operate as hermetically sealed units.

3.1.4 Electrical Performance

3.1.4.1 Purpose. The purpose of this test is to verify electrical performance of the component during the design qualification environmental test program. Satisfactory electrical performance before, during, and after the specified environments is required prior to approval of component design for incorporation in spacecraft.

3.1.4.2 Times of Performance. An initial test shall be conducted prior to the environmental tests to determine if performance meets the requirements of the particular spacecraft specification.

The test shall then be repeated before, during, and after each exposure in the temperature, humidity, vibration, acceleration, and thermal-vacuum tests (as specified in the provisions for these tests) to determine if those exposures adversely affect performance.

3.1.4.3 Initial Test

(a) Levels. This test requires application of expected parameters of voltage, impedance, and current as well as expected pulse timing and waveform at the electrical interfaces of the component. These parameters shall be varied throughout the parameter ranges specified by the interface document for the component. When practicable, the limits of parameter ranges shall be exceeded to determine the points beyond which the component ceases to perform.

In addition, each of the above electrical parameters shall be varied in a manner to approximate the sequence and levels in all normal modes of flight operation.

(b) Documentation. A record shall be made of all data necessary to determine that component performance meets the requirements of the particular spacecraft specification. These data provide a basis for determining satisfactory performance in the subsequent performance tests conducted before, during, and after the environmental exposures.

(c) Conditions. This test shall be conducted under standard conditions as defined in 1.9.4.

3.1.4.4 Succeeding Tests

(a) Levels. The parameters for the Initial Test (3.1.4.3a) shall apply to subsequent electrical performance tests specified in the design qualification test program except that the parameter range limits ordinarily shall not be exceeded.

(b) Documentation. A chronological log shall be maintained, indicating the duration of each operational period, the related environmental exposure, and the component configuration when appropriate.

(c) Conditions. Tests not conducted during environmental exposures shall be conducted under standard conditions as defined in 1.9.4.

3.1.4.5 Instrumentation. Sufficient interface instrumentation, actual or simulated, shall be used so that the inputs and outputs of the component under test can be measured and calibrated with conventional instruments.

Also precautions of 2.1.3.5 shall be used when applicable to particular components.

3.1.4.6 Test Procedures. The following techniques shall be employed in conduct of the electrical performance test:

(1) For each test, only one electrical parameter initially shall be varied at a time so that its net effect may be gaged accurately. Subsequently, realistic combinations of electrical parameters shall be applied to simulate adverse conditions.

(2) If power supplies supplant batteries during the test, the battery terminal impedance shall be simulated. If power supplies supplant solar paddles, the solar cell array impedance shall be simulated.

(3) If the component under test depends upon an external converter for multivoltage levels, the actual converter shall be used during the test.

(4) When feasible, experiments shall be activated by appropriate stimuli to check electrical performance.

3.1.5 Temperature and Humidity. The provisions of 2.1.8 for temperature and humidity testing of prototype spacecraft shall apply to components except that the operational test temperatures shall be 15°C above and below predicted orbital extremes as shown in Table XII. The electrical performance tests shall be conducted in accordance with 3.1.4.

TABLE XII

OPERATING TEMPERATURE TEST SCHEDULE
COMPONENT DESIGN QUALIFICATION

Cold Soak		Hot Soak	
Power-off Orbital Temperature	Operational Duration	Power-off Orbital Temperature	Operational Duration
15°C below prediction	Until temperature stabilized +6 hrs.	15°C above prediction	Until temperature stabilized +6 hrs.

3.1.6 <u>Structural Dynamics Phase</u>

3.1.6.1 Transportation and Handling. The provisions of 1.11 and 2.1.11.6 for handling and shipping spacecraft shall also apply to components.

3.1.6.2 Component Performance. Prior to and after the vibration test, the component shall be visually examined and functionally tested to check performance. During the test, the component shall be operated in a duty cycle typical of that to be employed in the launch phase and monitored for malfunctions.

Exact monitoring requirements shall be specified in the test plan for the particular spacecraft test program. The electrical performance testing shall be conducted in accordance with 3.1.4.

3.1.6.3 Components for Proto-Flight Spacecraft. (For definition of proto-flight spacecraft see 4.11.) Proto-flight components shall be subjected to random and sinusoidal vibration tests at design qualification levels but at flight acceptance durations.

3.1.6.4 Selection of Method

(a) Options. Either of the two methods given below may be used to qualify prototype components prior to integration with the prototype spacecraft. The tests are intended to approximate conditions experienced by the component when mounted in the spacecraft. Method I will serve this purpose in most cases; however, some components may require the more exact Method II tests (conducted with component attached to representative spacecraft structure) to avoid unrealistic failures. This is particularly true where exact simulation of boundary conditions is critical to the success of the test. The method to be used for each component should be specified in the particular spacecraft test specification.

(b) Selection Considerations. In general, the basic version of Method I will be found more economical, although somewhat less realistic than the optimal tests in Method II.

On the basis of practicability or where an undertest would result, Method I is recommended for all components except those where it is reasonably apparent that unrealistic failures necessitating expensive or unwarranted redesign would result. This would apply, for example, to components with critical boundary conditions or components which are necessarily fragile in order to meet performance requirements. An example of the former might be a deployable appendage; an example of the latter might be an experiment package using special vacuum tubes, or membranes, or having critical optical alignment requirements. An undertest might result, for example, when a flexible boom is mounted to a flexible panel which rotates as well as translates. In every case where base Method I is deviated from, reasonable justification should exist that the selected option is more realistic in the specific case in question; not simply that the test is easier.

The specific method and options to be used for each component should be specified in the detailed spacecraft specifications. It is urged that careful consideration be given to the selections; and it is recommended that the project office adopt a policy of extreme reluctance toward changing options, once the test program has begun.

3.1.6.5 Method I. In this method, the component is attached to the vibration exciter by means of a rigid fixture. The mounting should simulate, insofar as practicable, the mounting in the spacecraft. In particular, mounting brackets, vibration isolators (if any) and exact hardware should be used.

(a) Sinusoidal Vibration. The component, mounted as described above, shall be subjected to swept sinusoidal vibration along each of three mutually perpendicular axes and over a frequency range of 10 to 2000 Hz. The sweep rate shall be 2 octaves/minute. The levels from 10-200 Hz shall be based on unnotched spacecraft design qualification levels multiplied by measured or estimated transmissibility at the attachment location of the component. The level from 200-2000 Hz shall be 5 g-peak.

In cases where unnotched spacecraft inputs result in overly severe levels in the notched frequency band, the levels should be reevaluated, making use of calculated flight responses.

(b) Random Vibration. The component shall be subjected to a random vibration test in accordance with Table 3 of the applicable launch vehicle appendix. If desired, the spectra may be shaped in accordance with measured or estimated transmissibilities.

(c) Options

(1) Shock. In lieu of the 200-2000 Hz portion of the sinusoidal vibration test, a shock test based on the spacecraft shock test may be performed. The shock test should be realistically derived on the basis of the spectrum measured or estimated at the location of the component during the spacecraft shock test. (Note: because of the lack of experience in this area, it is expected

that a law of diminishing returns will be evident in that the effort required to increase the accuracy of the simulation may soon exceed any benefits accrued from such an increase. In most cases, the use of Method II will be found more efficient than this option).

(2) Acoustic Noise. An acoustic noise test may be substituted for the random vibration test for those components in which the acceleration responses are due mainly to direct acoustic inputs rather than to structure-borne inputs. An example is a component having a large surface area-to-mass ratio. In this case, the component is not rigidly attached, but is freely suspended in an acoustic test chamber selected to have field characteristics (progressive wave, semireverberant, etc.) most like the field seen by the component when mounted in the spacecraft. The acoustic levels are those specified for the prototype spacecraft modified by measured or estimated transmission characteristics.

3.1.6.6 Method II. In this method, components shall be correctly mounted (location, orientation, fastening, etc.) in a representative spacecraft structure. Additional substitute masses shall be assembled to the spacecraft structure as required to obtain a realistic dynamic simulation of the flight spacecraft.

The component/spacecraft assembly is then subjected to the tests specified in 2.1.9, 2.1.10 and 2.1.11 as applicable for the same options selected for the prototype spacecraft tests.

3.1.7 <u>Acceleration, Steady State</u>

3.1.7.1 Levels. Components shall be subjected to an acceleration test of 1 minute duration, specified in Table XIII.

3.1.7.2 Apogee/Retro Motor Spacecraft. Components on this type of spacecraft may require additional tests according to whether the spacecraft is spin-stabilized and according to direction and magnitude of motor thrust.

(a) Non-Spin Stabilized Spacecraft. If the retro/apogee motor thrust is in the same direction as the launch vehicle thrust, an acceleration test based on the greater of the two thrusts shall be required. If the retro/apogee motor thrust is in the opposite direction to the launch vehicle acceleration, an additional test simulating this acceleration shall be prescribed in the particular spacecraft specification.

(b) Spin-Stabilized Spacecraft. Components on this type of spacecraft shall require a test based on the most severe combination of spin and apogee thrust acceleration. An additional test based on the maximum launch vehicle thrust shall be required if this acceleration is significantly greater than the above combined acceleration or in the opposite direction to the apogee/retro motor thrust.

3.1.7.3 Component Performance. The provisions of 3.1.6.2 for checking performance shall also apply to acceleration testing.

3.1.7.4 Tolerances. The acceleration gradient imposed by the centrifuge across the component being tested shall not cause the specified acceleration to vary by more than ±10%. This means that the distance from the centrifuge pivot to the outermost edge of the component must be at least five times the distance across the component, measured in a radial direction from the centrifuge pivot.

3.1.7.5 Test Setup. Sealed units shall be pressurized to 760 torr (15 psi) in excess of prelaunch pressure in cases where changes in internal or external pressure may result in significant changes in strength, stiffness, applied loads or leak properties.

The component shall be attached to a mounting fixture in a manner which simulates the actual mounting of a component on the spacecraft structure.† The mounting, applied loads fixture shall be attached to the centrifuge so that the component may have acceleration applied in the direction of the resultant "(A)" of the acceleration due to thrust and spin (see Figure V).

†If the orientation of the component is unknown, the acceleration level determined from Table XIII shall be applied along each of three mutually orthogonal axes of the component in both directions, a total of six tests. If the location of the component is unknown, it shall be assumed to be mounted at an extreme radius for the purpose of calculating spin acceleration.

TABLE XIII

DESIGN QUALIFICATION COMPONENT ACCELERATION

Direction*	Acceleration** (Magnitude in g's)	Duration
Resultant of spin and thrust acceleration per 3.1.7.4	$\sqrt{A_t^2 + A_r^2}$	1 minute

*See Figure V for spacecraft coordinate system.

**A_t = Required test level in g's for prototype spacecraft as calculated in Table 5 of the appropriate launch vehicle appendix (1.5 times maximum expected spacecraft acceleration).

A_r = $.0000426\ r\ N^2$ = 1.5 times radial acceleration from final stage spin where:

r = radius in inches from the spin axis of the spacecraft to the CG of the component

N = maximum spin rate expected in final stage flight in rpm.

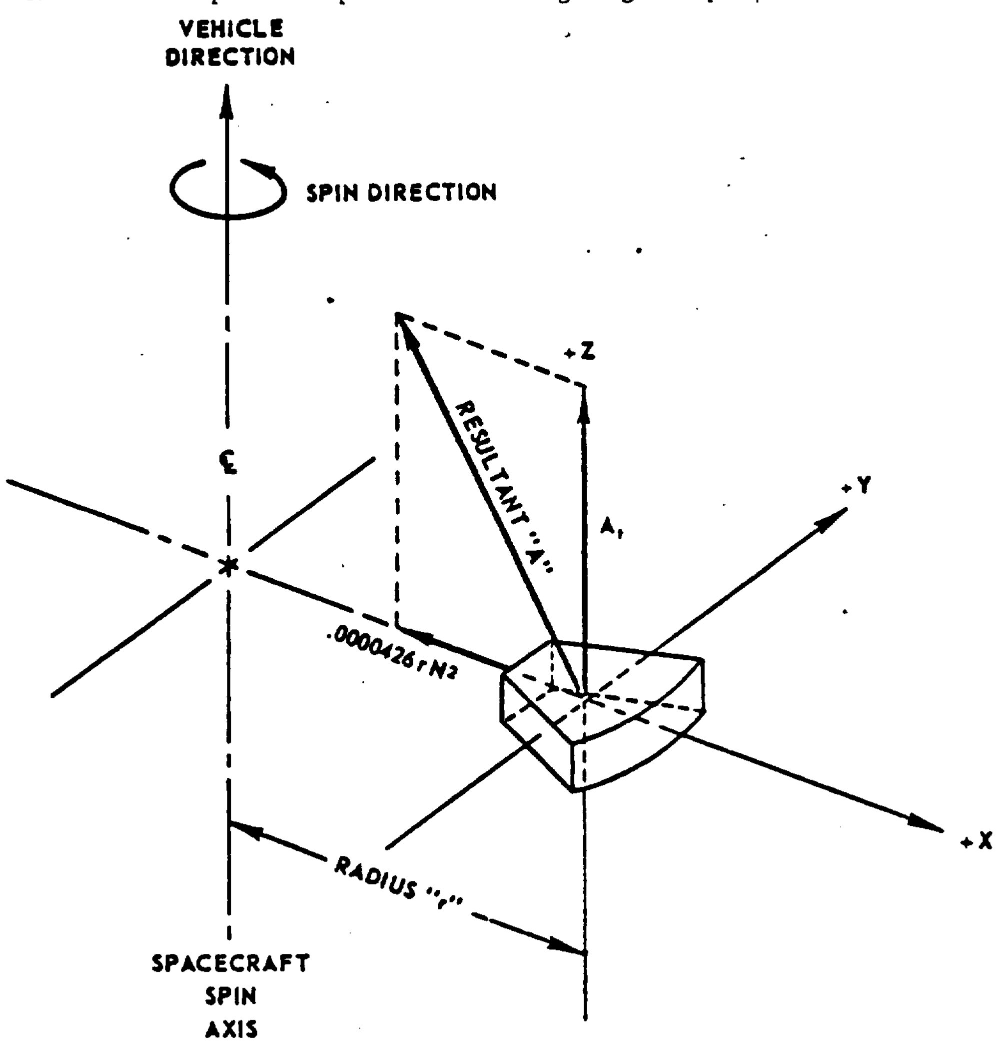

Figure V–Component/Subsystem/Experiment Coordinate System

3.1.8 Thermal-Vacuum

3.1.8.1 General Requirements. The purpose of this test phase is to prove the component design by checking its performance capability under vacuum and temperature stress more severe than predicted for orbit. The temperature extremes shall be 10°C above the maximum and 10°C below the minimum temperatures predicted for orbit (see paragraph 2.1.13.4). A corona check shall be performed as the initial portion of the test to verify that no significant electrical discharge occurs during transition to vacuum conditions.

During the test, care shall be taken that the maximum rate of temperature change does not exceed acceptable limits, based on component characteristics or orbital predictions, whichever are more severe.

The provisions of 3.1.4 for electrical performance testing shall apply.

3.1.8.2 Test Setup. The provisions of 2.1.8.4 shall apply.

3.1.8.3 Monitoring. The temperature of the component shall be monitored continuously during the test. Performance of the component shall be monitored at least at the beginning and end of each soak period. For cyclically operated components, the performance of the component shall be checked periodically, consistent with the operating duty cycle required.

3.1.8.4 Corona. The instrumented components shall be placed in the test chamber, and an operational check shall be performed prior to chamber evacuation. With the component in operational status for launch, the chamber shall be evacuated to 1×10^{-4} torr. Corona and other high voltage effects shall be monitored throughout the evacuation period.

3.1.8.5 Plasma Test. A plasma test shall be conducted immediately following the corona check in accordance with 3.1.11 if indicated by component characteristics.

3.1.8.6 High Temperature Soak. Immediately following the corona check or plasma test, the component shall be turned off and the chamber evacuated until a pressure of less than 1×10^{-5} torr is achieved in the chamber working volume. The wall temperature shall be increased during this time to cause the component to stabilize at 10°C above the maximum predicted orbital temperatures. Stabilization shall be assumed when indication from the control thermocouple attached to the component does not change by more than 0.5°C per hour.

When stable conditions have been achieved, the chamber temperature controls shall be held fixed. The component shall be operated and its temperature monitored throughout the exposure period of 24 hours.

3.1.8.7 Low Temperature Soak. At the conclusion of the high temperature vacuum exposure the chamber conditions shall be altered such that stable conditions are established at 1×10^{-5} torr and 10°C below the minimum predicted temperature. Stabilization shall be assumed when indication from the control thermocouple attached to the component does not change by more than 0.5°C per hour.

When stable conditions have been achieved, the chamber temperature controls shall be held fixed. The component shall be operated and its temperature monitored throughout the exposure period of 24 hours.

For cyclically operated components ("on-off" orbital operation), cold start capability shall be demonstrated at least three times during the 24-hour exposure. Each cycle of operation shall be preceded by reverting to stabilized conditions at 10°C below the minimum orbital temperature.

3.1.9 Electromagnetic Interference. The appropriate provisions for EMI testing of prototype spacecraft, 2.1.15, shall apply to components.

3.1.10 Final Magnetic Field Measurement. After completion of the specified environmental test program, components shall be subjected to a magnetic field measurement as required by the particular spacecraft specification to determine the effects of such tests on magnetic moments of the components. The component shall be positioned inside the magnetic test coils via a fixture of nonmagnetic material. If necessary, the component shall be subjected to a deperm treatment to reduce such moments to within the limits previously measured in 3.1.2.

3.1.11 Plasma

3.1.11.1 Background. Orbital failures of spacecraft components have occurred which were not apparent in the ground test program. A contributing factor in the failures of some flight experiments carrying photomultipliers or electron multipliers was felt to be the charged particles of thermal energies in the orbital environment. The densities of these particles, (0+, H+, and electrons) reach maximums on the order of 10^6 per cubic centimeter. The motion of the spacecraft through the ions and the spacecraft potential can give the ions relative energies up to 25 electron volts. These particles can initiate breakdown in high voltage circuits or scintillation and false counting in open cathode photomultiplier tubes.

3.1.11.2 Determination of Test Requirement. If a component/subsystem/experiment has high voltage or other sensitive circuits that will be exposed to the orbital plasma environment, a plasma test shall be conducted to subject the component to a simulation of the charged particle environment expected.

3.1.11.3 Time of Performance. Ordinarily, the plasma test shall be performed during the thermal-vacuum test sequence following the corona check (3.1.8.4). When the chamber has been evacuated to a pressure of less than 5×10^{-5} torr, the plasma environment will be established and its characteristics analyzed by Faraday Cups and/or retarding potential analyzers. The energies and densities of the charged particles shall be determined by the intended orbital characteristics of the mission and the estimated spacecraft potential. A functional test of the component/subsystem/experiment in flight configuration shall then be performed in the plasma environment.

3.2 COMPONENT FLIGHT ACCEPTANCE

Flight acceptance testing of components has the purpose of locating latent material and workmanship defects in components of proven design before they are integrated into the flight spacecraft system or before they are accepted as flight spares.

The separate flight acceptance testing of components which are to be tested as part of an assembled flight spacecraft may be waived by the project manager

The provisions of this section apply to subsystems and experiments as well as to components.

3.2.1 Leak Detection. The provisions of 2.2.1 for flight spacecraft leak detection shall apply to components which operate as hermetically sealed units.

3.2.2 Electrical Performance

3.2.2.1 Purpose. The purpose of this test is to verify electrical performance during the component flight acceptance test program. Satisfactory electrical performance before, during and after the specified environments shall be required prior to integration with flight spacecraft.

3.2.2.2 Times of Performance. An initial test shall be conducted prior to the environmental tests to determine if component performance meets the requirements of the particular spacecraft specification.

The electrial performance test shall then be repeated before, during, and after each exposure in the vibration and thermal-vacuum tests (as specified in the provisions for those tests) to determine if the exposures adversely affect performance.

3.2.2.3 Initial Test

(a) Levels. The levels for this test shall be the same as for prototype components (3.1.4.3a) except that the parameter ranges specified by the component interface document shall not be exceeded.

(b) Documentation. The requirements shall be the same as for prototype components (3.1.4.3b).

(c) Conditions. This test shall be conducted under standard conditions as defined in 1.9.4.

3.2.2.4 Succeeding Tests. Same as for prototype components (3.1.4.4).

3.2.2.5 Instrumentation. Same as for prototype components (3.1.4.5).

3.2.2.6 Test Procedures. Same as for prototype components (3.1.4.6).

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3.2.3 Structural Dynamic Phase

3.2.3.1 General Requirements. Same as 3.1.6.

3.2.3.2 Flight Components. Flight components, which are tested later as part of a completely assembled flight spacecraft, shall be subjected to random vibration and, where applicable, to acoustic noise tests. Random vibration and acoustic noise levels are to be taken from Tables 8 and 9 of the applicable launch vehicle appendix. If desired, the spectra may be shaped in accordance with measured or estimated transmissibilities.

3.2.3.3 Flight Spare Components. Flight spare components that have not been exposed to system tests as part of a prototype or backup spacecraft shall receive both sinusoidal and random vibration. See 1.8.3 for a general discussion on the testing of flight spares.

Random vibration shall be the same as for flight components as stated in 3.2.3.2. Sinusoidal vibration from 10-200 Hz shall be based on unnotched spacecraft flight acceptance levels multiplied by measured or estimated transmissibilities. Levels from 200-2000 Hz shall be ±3.3 g. Sweep rates shall be the same as for spacecraft flight acceptance.

In cases where unnotched spacecraft inputs result in overly severe levels in the notched frequency band, the levels should be reevaluated making use of calculated flight responses.

3.2.4 Thermal-Vacuum

3.2.4.1 General Requirements. The purpose of this test phase is to locate latent material and workmanship defects in a component of proven design by checking its performance capability under vacuum at the temperature extremes expected in flight. A corona check shall be performed as the initial portion of the test to verify that no significant electrical discharge or degradation occurs during transition to vacuum conditions.

During the test, care shall be taken that the maximum rate of temperature change does not exceed acceptable limits, based on component characteristics or orbital predictions, whichever are more severe.

The provisions of 3.2.2 for electrical performance testing shall apply.

3.2.4.2 Other Requirements. The provisions of 3.1.8 for test setup, monitoring, corona, high temperature soak, and low temperature soak of prototype components shall apply to flight components except that the temperature extremes shall equal predicted minimum and maximum extremes instead of exceeding them by 10°C.

3.2.4.3 Duration of Flight Spares Test. The reliability assurance plan shall specify the test duration for each spare component and the duration shall be included in the particular spacecraft specification.

The minimum time is 24 hours "hot" and 24 hours "cold" as required for flight components. The maximum time would be that obtained when components are "run-in" for long periods under vacuum. Between these extremes, an appropriate duration may be selected from Table XIV after a careful consideration of the following factors:

(a) Spare Designation Uncertain. When a choice between the spare and the flight component is not made until late in the test program—which is frequently the case with experiments—both units should be given an extended test to assure sufficient exposure of the spare.

In such event, care must be taken to avoid an overtest of the flight unit which is later to be tested as part of the integrated spacecraft.

(b) Importance to Mission Success. A spare for an in-line, nonredundant component would obviously require more thorough testing than one less critical to mission success.

(c) Voltage Level of Component. Consideration should be given to the highest voltage level present in the component. High voltage circuits are prone to break down electrically at intermediate pressures between atmospheric and "hard" vacuum. Inadequate venting and "slow" leaks may prevent the reaching of intermediate vacuum conditions in a short test. However, rather than perform long tests on all high voltage circuits as a routine matter, a better approach is to review carefully the physical design of the spare and the use of hermetic seals, potting and venting with a view to a test of more moderate duration.

TABLE XIV

FLIGHT ACCEPTANCE
SPARE COMPONENT THERMAL-VACUUM

Spare Component Category	Test Duration
In-line, Nonredundant Mission Critical	Four and one-half cumulative days at highest predicted orbital temperature; four and one-half cumulative days at lowest predicted orbital temperature. At least eight transitions to extreme temperatures.
High Voltage >200^{v}	Three cumulative days at highest predicted orbital temperatures; three cumulative days at lowest predicted orbital temperature. Vacuum shall not be interrupted during the test.
All Other	One day at highest predicted orbital temperature; one day at lowest predicted temperature. Vacuum shall not be interrupted during the test.

3.2.5 Electromagnetic Interference. The appropriate EMI provisions for testing prototype spacecraft, 2.1.15, shall apply to flight components as well as to prototype components.

SECTION 4

DEFINITIONS

4.1 DESIGN QUALIFICATION ENVIRONMENTAL TESTS. These are tests intended to simulate conditions more severe than those actually expected from ground handling, launch, and orbital operations in order to provide assurance of discovering any design deficiencies which might exist. They are not intended to exceed design safety margins and care shall be taken not to introduce unrealistic modes of failure. In the case of ground handling, control of the shipping and handling modes and the use of properly designed shipping containers shall prevent the imposition of unnecessarily high design margins.

4.2 FLIGHT ACCEPTANCE ENVIRONMENTAL TESTS. These tests are intended to simulate actual conditions expected from ground handling, launch, and orbital operations. The purpose is to locate latent material and workmanship defects in a proven design.

4.3 EXPERIMENT. A combination of a sensor and associated electronics borne by a spacecraft and having the purpose of collecting data for space research.

4.4 COMPONENT. A self-contained combination of parts and/or assemblies within a subsystem performing a function necessary to subsystem operation. Examples are: receivers, transmitters, modulators.

4.5 SUBSYSTEM. A functioning entity consisting of two or more components within a major system (launch vehicle, spacecraft, or ground support); such as propulsion subsystem of a launch vehicle or attitude control subsystem of a spacecraft.

4.6 SYSTEM. May be defined as follows:

(a) Used in the term, "space system," comprising the launch vehicle, spacecraft, and ground support equipment for launch and flight.

(b) One of the major subdivisions of a space system, such as launch vehicle, spacecraft, or ground support system.

(c) May have same meaning as subsystem. See definition of subsystem.

4.7 SPACECRAFT. The complete assembly of components, experiments, subsystems, and structure with associated interconnecting cabling, designed to be placed into an orbit about the earth or into a trajectory to another celestial body.

4.8 STRUCTURAL MODEL. The structural model of the spacecraft should duplicate the framework of the spacecraft as to material, configuration, weight, and weight distribution. Representation of the loaded spacecraft for structural model testing is achieved by mounting dummy weights to simulate components, subsystems, and experiments and their distribution. The spacecraft structural model is subjected to static load tests, spin, acceleration, vibration, and mechanical functions at design qualification levels to establish confidence in the structural design prior to design qualification testing of the prototype spacecraft.

4.9 PROTOTYPE. Refers to an operating spacecraft, subsystem, component, or experiment which is subjected to design qualification environmental tests to qualify a design for fabrication of flight model equivalents.

4.10 FLIGHT. Describes spacecraft, component, subsystem, or experiment which is to be used operationally and thus is subject to flight acceptance environmental tests. Successful completion of these tests qualifies the spacecraft for launch and the component/subsystem/experiment for incorporation into the flight spacecraft.

4.11 PROTO-FLIGHT SPACECRAFT. This term is used in two ways:

(a) A spacecraft that had originally been designated as a prototype and subjected to complete or partial design qualification environmental testing and which is redesignated for flight operations. The prototype testing may have included only the structure, or the structure plus any or all of the components/subsystems/experiments. The provisions for valid test programs for such proto-flight spacecraft vary widely and should be carefully considered on a case by case basis. The particular spacecraft specification should not necessarily adhere closely to the provisions for spacecraft flight acceptance testing contained in this document but should take into account prototype testing already accomplished to avoid overtesting.

(b) A spacecraft which is designated in advance to serve both as a prototype and flight model and thus should be subjected to environmental tests which combine elements of flight acceptance and prototype testing; that is by applying design qualification levels and flight acceptance durations.

Paragraphs 2.1.16.6 a and b contain some recommendations for conducting the operational and mechanical functioning phase of a proto-flight spacecraft test program.

4.12 GENERAL ENVIRONMENTAL TEST SPECIFICATION. This document stipulates basic environmental test levels for all spacecraft using a given class of launch vehicles. It serves as the principal source and as a model in form for writing the particular environmental test specifications required for each spacecraft program.

4.13 PARTICULAR ENVIRONMENTAL TEST SPECIFICATION. This document stipulates test levels, conditions, and tolerances on measurements for the environmental test program of a particular spacecraft. It uses the General Environmental Test Specification as its principal source document but takes into account special characteristics of the particular spacecraft.

4.14 TEST PLAN. This document governs the conduct of the environmental test program for a particular spacecraft. It provides for the execution of the Particular Environmental Test Specification by stipulating the test sequence and schedule, spacecraft configuration for each exposure, variables to be measured, and requirements for facilities and personnel.

4.15 AXIS Z-Z THRUST*

(a) Spacecraft. The Z-Z axis shall means a line, passing through the center of mass of the spacecraft, that is parallel to the thrust axis of the launch vehicle. Positive Z axis direction shall be in the direction of launch vehicle flight.

(b) Component. The Z-Z axis shall mean a line passing through the center of mass of the installed component that is parallel to the thrust axis of the launch vehicle. Positive Z axis direction shall be in the direction of launch vehicle flight.

*See Figure V.

4.16 AXIS X-X LATERAL*

(a) Spacecraft. The X-X axis shall mean an arbitrarily selected line, passing through the center of mass of the spacecraft, that is perpendicular to the Z-Z axis. Positive X axis direction shall be selected in accordance with the right-hand rule.

(b) Component. The X-X axis shall mean a line passing through the center of mass of the installed component that is parallel to the lateral (X-X) axis of the spacecraft. Positive X axis direction shall be the same as positive spacecraft X axis direction.

4.17 AXIS Y-Y LATERAL*

(a) Spacecraft. The Y-Y axis shall mean a line passing through the center of mass of the spacecraft that is mutually orthogonal with the X-X and Z-Z axes. Positive Y axis direction shall by selected in accordance with the right-hand rule.

(b) Component. The Y-Y axis shall mean a line passing through the center of mass of the installed component that is mutually orthogonal with the X-X and Z-Z axes. Positive Y axis direction shall be the same as positive spacecraft Y axis direction.

4.18 SPARE (OR FLIGHT SPARE). A component which has been designated to serve on a flight model spacecraft as a replacement for a component of identical design should the installed component be damaged or fail. When a spare is selected as a replacement, its origin and test history is a major consideration. It must have undergone a thorough test as part of a prototype or backup spacecraft or have been separately subjected to appropriate flight acceptance tests for components.

4.19 BASE STRAIN SENSITIVITY. An undesirable characteristic of certain piezoelectric accelerometers in which spurious outputs are produced when curvature and strain are introduced to the base of the accelerometer by the surface to which it is attached.

4.20 NOTCHING. The reduction of sinusoidal vibration test levels in a resonant frequency band of a spacecraft to prevent the application of unrealistic loads.

*See Figure V.

APPENDIX A — DELTA L, M, N

SPACECRAFT DESIGN QUALIFICATION

To Be Used in Conjunction with Procedures in Section 2.1 As Shown by the Paragraph Numbers Preceding Each Table or Figure.

This section of Appendix A is restricted to the special test levels and procedures that are to be used for the Design Qualification of Spacecraft that are to be launched by Delta vehicles L, M, and N as described below.

DELTA MODELS — GSFC DESIGNATIONS

Delta Model	First Stage	Thrust Augmenters	Second Stage	Third Stage
L	Long Tank Thor	3 Solid Strap-On	Improved Delta	FW-4
M	Long Tank Thor	3 Solid Strap-On	Improved Delta	TE 364-3
N	Long Tank Thor	3 Solid Strap-On	Improved Delta	None

Other, older Delta models that have limited availability such as DSV-3E-1, 3G, or 3J are to be considered on a case by case basis.

Balance 2.1.5

TABLE A-1

SPACECRAFT DESIGN QUALIFICATION
DELTA BALANCE REQUIREMENTS

Delta Model	Configuration	Static Unbalance (gm-in.)	Dynamic Unbalance (gm in.^{2})
L and M	Launch*	$<0.03 \times W$**	$<800 \times I_s$***
N	Launch*	Delta Project Office, GSFC	Delta Project Office, GSFC
All Models	Orbital	Per Particular Spacecraft Specification	Per Particular Spacecraft Specification

*Based on Delta Design Restraints Manual, January 1966 (SM-48897)
**W = Spacecraft weight (grams).
***I_s = Spacecraft spin axis moment of inertia (slug ft.^{2})

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Vibration 2.1.9

Applicability of Levels. The vibration levels specified in Table A-2 and A-3 apply to spacecraft which are to be launched from a wet deflector-plate type launch pad using a standard Delta DSV#3E-7 spacecraft fairing. Exceptions will require a review of test levels for adequacy. Levels for the Delta N (2 stage) vehicle do not include the effects of a spintable.

TABLE A-2

SPACECRAFT DESIGN QUALIFICATION
DELTA SINUSOIDAL VIBRATION

Delta Model	Axis*	Frequency Range (Hz)	Level "g" (0-to-Peak)	Sweep (Octaves/Minute)
L (3 Stage)	Thrust (Z-Z)	5-11	.48 in. d.a.	2.0
		11-17	±3.0g	2.0
		17-23	±7.5g	1.5
		23-200	±2.3g	2.0
	Lateral (X-X and Y-Y)	5-7.5	.80 in. d.a.	2.0
		7.5-13	±2.3g	2.0
		13-200	±1.5g	2.0
M (3 Stage)	Thrust (Z-Z)	5-11	.48 in. d.a.	2.0
		11-17	±3.0g	2.0
		17-23	±6.0g	1.5
		23-200	±2.3g	2.0
	Lateral (X-X and Y-Y)	5-7.5	0.8 in. d.a.	2.0
		7.5-13	±2.3g	2.0
		13-200	±1.5g	2.0
N (2 Stage)	Thrust (Z-Z)	5-11	.48 in. d.a.	2.0
		11-17	±3.0g	2.0
		17-23	±4.5g	1.5
		23-200	±2.3g	2.0
	Lateral (X-X and Y-Y)	5-7.5	0.53 in. d.a.	2.0
		7.5-200	±1.5g	2.0

*See 4.15, 4.16, and 4.17 for definition; see also Figure V.

Vibration 2.1.9

TABLE A-3

SPACECRAFT DESIGN QUALIFICATION
DELTA RANDOM VIBRATION

Delta Model	Axis*	Frequency Range (Hz)	PSD Level (g^2/Hz)	Acceleration (g-rms)	Duration
L and M (3 Stage)	X-X, Y-Y and Z-Z	20-300	.0029 to .045 Increasing from 20 Hz at rate of 3 db/ octave	9.1	4 minutes each axis
		300-2000**	.045		
N (2 Stage)	X-X, Y-Y and Z-Z	20-300	.0023 to .090 Increasing from 20 Hz at rate of 3 db/ octave	12.8	4 minutes each axis
		300-2000**	.090		

*See 4.15, 4.16, and 4.17 for definition; see also Figure V.
**The filter roll-off characteristic above 2000 Hz shall be at a rate of 40 db/octave or greater.

Acoustic Noise 2.1.10

The levels specified in Table A-4 apply to spacecraft which are to be launched from the Eastern Test Range, Pad 36-A and 36-B, using a 120-inch spacecraft fairing.

TABLE A-4

SPACECRAFT DESIGN QUALIFICATION
DELTA ACOUSTIC NOISE

Octave Band Center Frequency (Hz)	Sound Pressure Level (Db, ref. 0.0002 Microbar)		Duration
	Exterior of Shroud	Interior of Shroud	
15.8	139	127	2.0 minute for complete exposure
31.5	140	127	
63	144	129	
125	146	134	
250	149	137	
500	149	138	
1000	147	138	
2000	144	135	
4000	140	131	
8000	135	125	
Overall	155	144	

Shock 2.1.11 Apply the appropriate shock spectrum twice along the three major axes of the spacecraft.

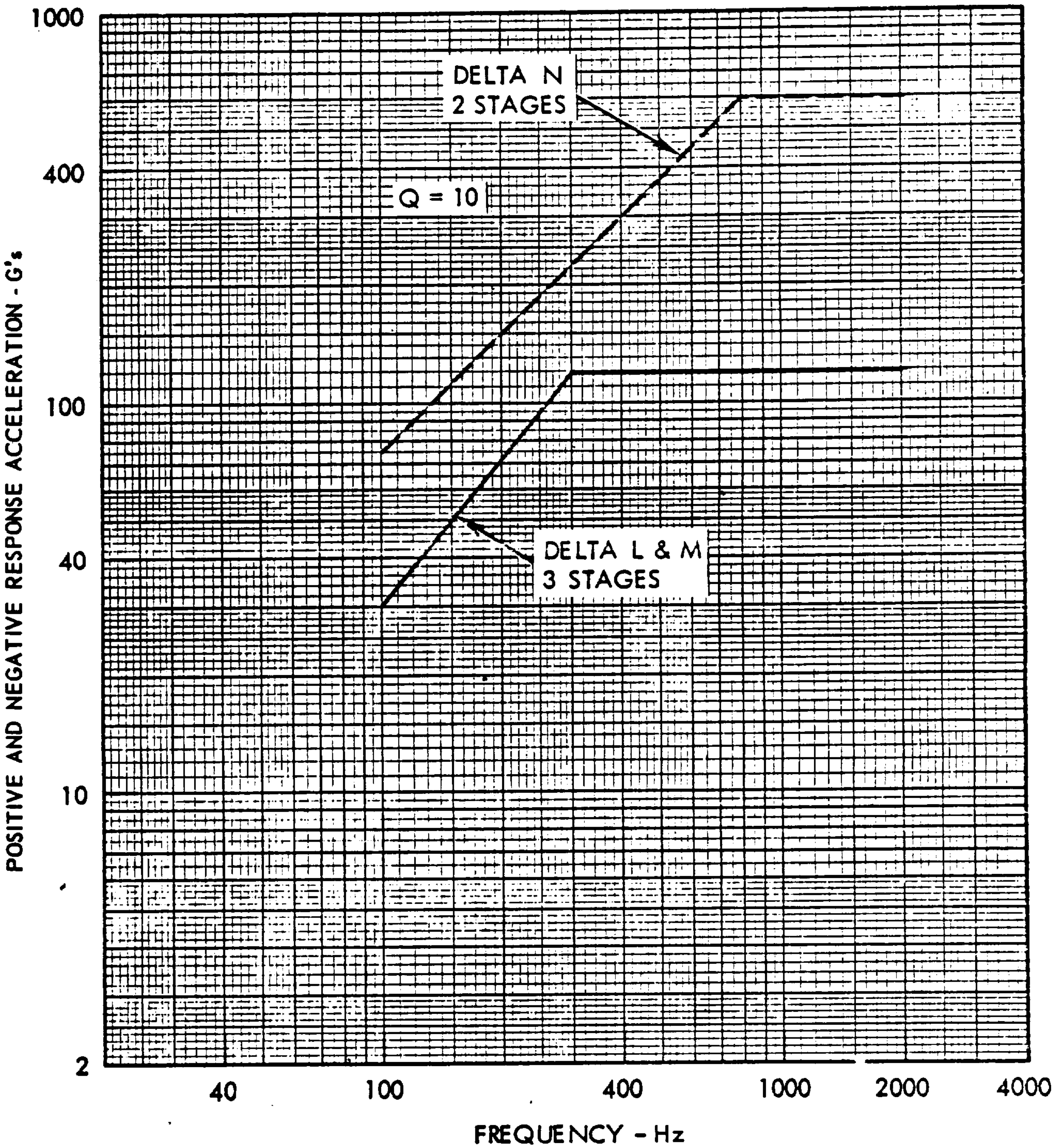

Figure 1-A. Delta Shock Spectra Inputs - Spacecraft Design Qualification

Structural Loads and Acceleration, Steady State (4.1.13)

TABLE A-5

SPACECRAFT DESIGN QUALIFICATION
DELTA ACCELERATION TEST LEVELS

Delta Model	Simultaneously Applied		Thrust Only	Duration
	1st Stage Thrust	Acceleration Lateral	3rd Stage Acceleration	
Delta L (FW-4 3rd Stage)	16.5	3.0	1.5 x A*	One Minute
Delta M (TE 364-3 3rd Stage)	14.7	3.0	1.5 x A*	
Delta N (No 3rd Stage)	13.8	3.0	None	

*Acceleration read from the applicable curve of Figure 2-A, Third Stage Acceleration vs Weight. If the third stage acceleration is more than 10% greater than the first stage acceleration, an additional one-minute thrust axis test is required.

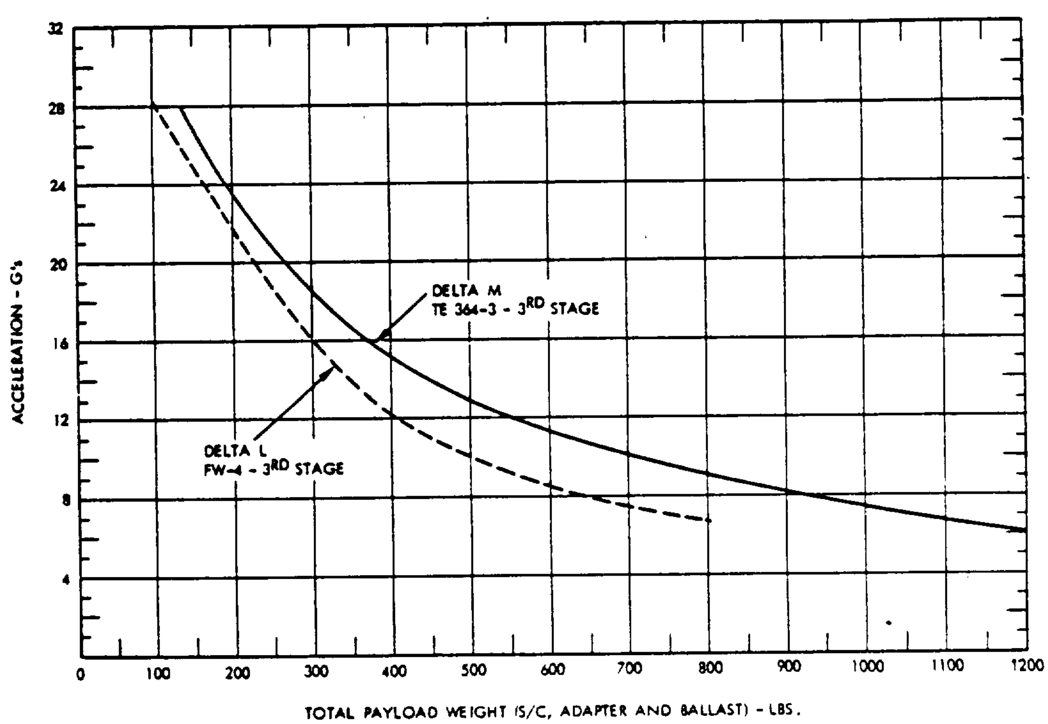

Figure 2-A. Delta Third Stage - Acceleration Vs Weight

APPENDIX A - DELTA L, M, N

SPACECRAFT FLIGHT ACCEPTANCE

To Be Used in Conjunction with Procedures in Section 2.2 as Shown by the Paragraph Numbers Preceding Each Table or Figure.

This section of Appendix A is restricted to the special test levels and procedures that are to be used for the Flight Acceptance of Spacecraft that are to be launched by Delta vehicles L, M, and N as shown below.

DELTA MODELS — GSFC DESIGNATIONS

Delta Model	First Stage	Thrust Augmenters	Second Stage	Third Stage
L	Long Tank Thor	3 Solid Strap-On	Improved Delta	FW-4
M	Long Tank Thor	3 Solid Strap-On	Improved Delta	TE 364-3
N	Long Tank Thor	3 Solid Strap-On	Improved Delta	None

Other, older Delta models that have limited availability such as DSV-3E-1, 3G, or 3J are to be considered on a case by case basis.

Balance - Initial 2.2.4

TABLE A-6

SPACECRAFT FLIGHT ACCEPTANCE
DELTA INITIAL BALANCE REQUIREMENTS

Delta Model	Configuration	Static Unbalance (gm-in.)	Dynamic Unbalance (gm in.^{2})
L and M	Launch*	$0.0225 \times W$**	$600 \times I_s$***
N		Per Delta Project Office, GSFC	Per Delta Project Office, GSFC
All Models	Orbital	Per Particular Spacecraft Specification	Per Particular Spacecraft Specification

*Based on Delta Design Restraints Manual, January 1966 (SM-48897).
**W =Spacecraft Weight (grams).
***I_s =Spacecraft spin axis Moment of Inertia (slug ft.^{2}).

Vibration 2.2.7

Applicability of Levels. The vibration levels specified in Table A-7 and A-8 apply to spacecraft which are to be launched from a wet deflector-plate type launch pad using a standard Delta DSV#3E-7 spacecraft fairing. Levels for the Delta N (2 stage) vehicle do not include the effects of a spintable. Exceptions will require a review of test levels for adequacy.

TABLE A-7

SPACECRAFT FLIGHT ACCEPTANCE
DELTA SINUSOIDAL VIBRATION

Delta Model	Axis*	Frequence Range (Hz)	Level "g" (0-to-Peak)	Sweep (Octaves/Minute)
L (3 Stage)	Thrust (Z-Z)	5-11	0.32 in. d.a.	4.0
		11-17	±2.0g	4.0
		17-23	±5.0g	3.0
		23-200	±1.5g	4.0
	Lateral (X-X and Y-Y)	5-7.5	0.52 in. d.a.	
				4.0
		7.5-13	±1.5g	4.0
		13-200	±1.0g	4.0
M (3 Stage)	Thrust (Z-Z)	5-11	0.32 in. d.a.	4.0
		11-17	±2.0g	4.0
		17-23	±4.0g	3.0
		23-200	±1.5g	4.0
	Lateral (X-X and Y-Y)	5-7.5	0.52 in. d.a.	4.0
		7.5-13	±1.5g	4.0
		13-200	±1.0g	4.0
N (2 Stage)	Thrust (Z-Z)	5-11	0.32 in. d.a.	4.0
		11-17	±2.0g	4.0
		17-23	±3.0g	3.0
		23-200	±1.5g	4.0
	Lateral (X-X and Y-Y)	5-7.5	0.34 in. d.a.	4.0
		7.5-200	±1.0g	4.0

*See 4.15, 4.16, and 4.17 for definition; see also Figure V.

Vibration 2.2.7

TABLE A-8

SPACECRAFT FLIGHT ACCEPTANCE
DELTA RANDOM VIBRATION

Delta Model	Axis*	Frequency Range (Hz)	PSD Level (g^2/Hz)	Acceleration (g-rms)	Duration
L and M (3 stage)	X-X, Y-Y and Z-Z	20-300	.0013 to .02 Increasing from 20 Hz at rate of 3 db/octave	6.1	2 minutes each axis
		300-2000**	.02		
N (2 stage)	X-X, Y-Y and Z-Z	20-300	.0010 to .04 Increasing from 20 Hz at rate of 4 db/octave	8.5	2 minutes each axis
		300-2000**	.04		

*See 4.15, 4.16, and 4.17 for definition; see also Figure V.
**The filter roll-off characteristics above 2000 Hz shall be at a rate of 40db/octave or greater.

Acoustic Noise 2.2.8

The levels specified in Table A-9 apply to spacecraft which are to be launched from the Eastern Test Range, Pad 36-A and 36-B, using a 120-inch spacecraft fairing.

TABLE A-9

SPACECRAFT FLIGHT ACCEPTANCE
DELTA ACOUSTIC NOISE

Octave Band Center Frequency (Hz)	Sound Pressure Level (db, ref. 0.0002)		Duration
	Exterior of Shroud	Interior of Shroud	
15.8	135	123	1 minute for complete exposure
31.5	136	123	
63	140	125	
125	142	130	
250	145	133	
500	145	134	
1000	143	134	
2000	140	131	
4000	136	127	
8000	131	121	
Overall	151	140	

Shock 2.2.9 Apply the appropriate shock spectrum once along the three major axes of the spacecraft.

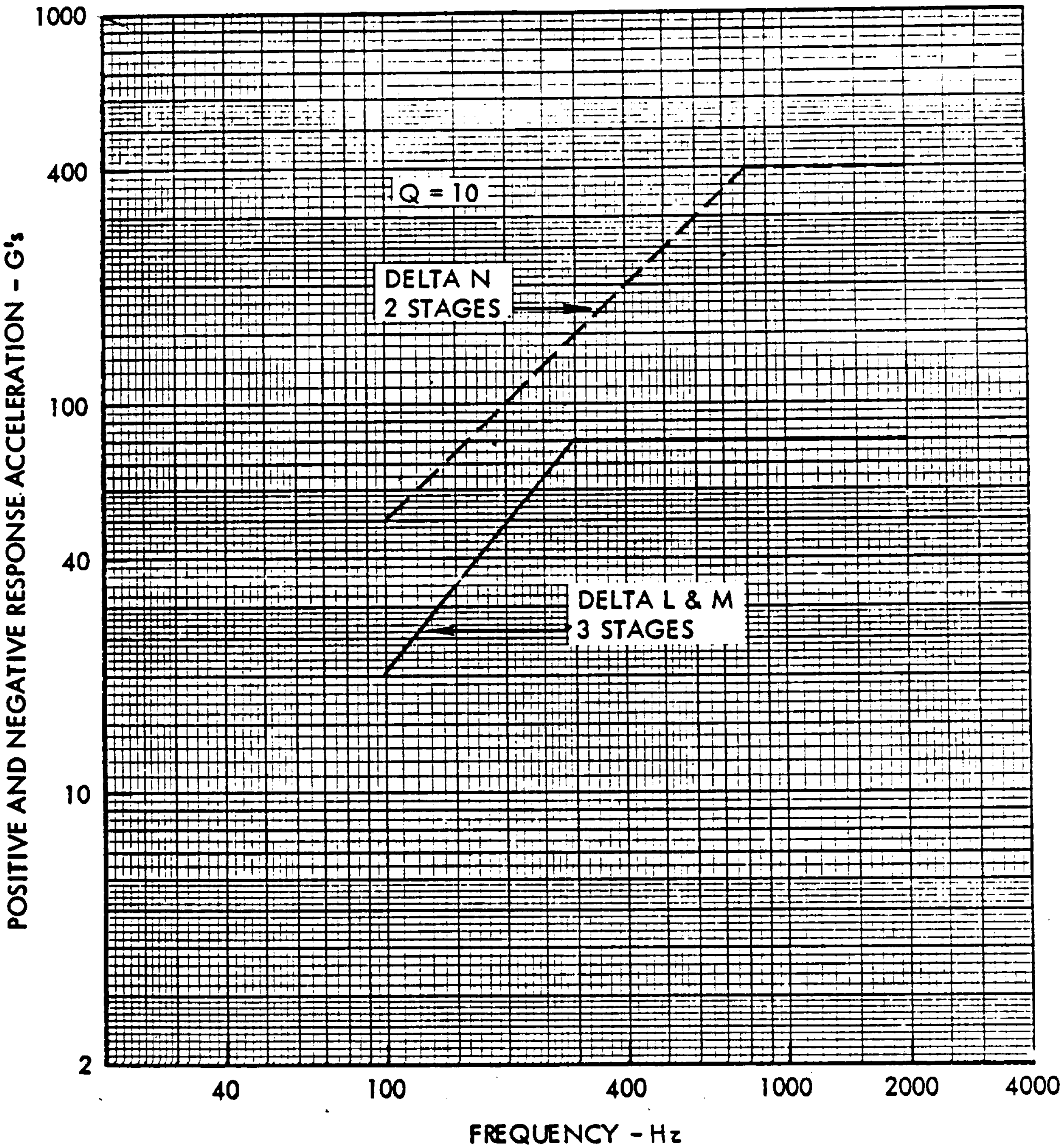

Figure 3-A. Delta Shock Spectra Input - Spacecraft Flight Acceptance

Final Balance 2.2.11

TABLE A-10

SPACECRAFT FLIGHT ACCEPTANCE
DELTA FINAL BALANCE REQUIREMENTS

Delta Model	Configuration	Static Unbalance (gm-in.)	Dynamic Unbalance (gm in.^{2})
L and M	Launch*	0.015 × W**	400 × I_s***
N		Per Delta Project Office, GSFC	Per Delta Project Office, GSFC
All Models	Orbital	Per Particular Spacecraft Specification	Per Particular Spacecraft Specification

*Based on Delta Design Restraints Manual, January 1966 (SM-48897).
**W =Spacecraft Weight (grams).
***I_s =Spacecraft spin axis moment of inertial (slug ft.^{2}).